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# ITOS/Space Shuttle Study Report

Prepared for  
Goddard Space Flight Center  
National Aeronautics and Space Administration  
Washington, D.C.

March 30, 1971

NAS 5-21600

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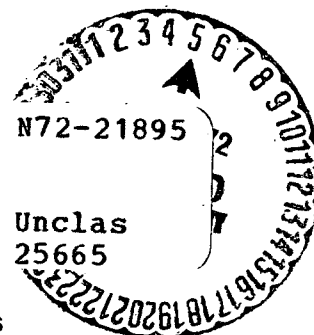
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## **PREFACE**

This report describes the results of a study to explore potential cost reductions that could be realized in the operational ITOS weather satellite program as a consequence of Shuttle/Tug availability. The study was conducted by the Astro-Electronics Division of RCA Corporation for the Goddard Space Flight Center of the National Aeronautics and Space Administration.

*NAS 5-21600*

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# SECTION I

## INTRODUCTION

### A. PURPOSE

The purpose of the ITOS/Space Shuttle Study was to explore potential cost reductions that could be realized in the operational ITOS weather satellite program as the result of Shuttle/Tug availability. As necessary background for the report, a brief description of the ITOS mission and a general definition of the space shuttle characteristics are presented.

#### 1. ITOS Mission

ITOS is a three-axis-stabilized earth-oriented spacecraft designed to provide full day and night global weather coverage on a daily basis. Television cameras furnish daytime picture coverage of the sunlit portion of the earth, while infrared radiometers, sensitive to surface temperatures of the earth, sea, and cloud tops provide both daytime and nighttime coverage.

The ITOS orbit is circular and near-polar with an altitude of 790 nautical miles and an angle-of-inclination of 102 degrees. The total orbital period is approximately 115 minutes (67 minutes in sunlight and 48 minutes in earth shadow). The earth rotates beneath the orbit 28.8 degrees during this period, allowing the satellite to observe a different portion of the earth's surface with sufficient overlap from orbit to orbit.

The orbit is sun-synchronous and precesses eastward about the earth's polar axis at 1 degree per day, or at the same rate and direction as the earth's average annual revolution about the sun. The present ITOS-1 launched into an ascending node orbit, will always cross the equator at 3 p.m. northbound and 3 a.m. southbound local time.\* Sun-synchronous orbits compensate for seasonal variations by keeping the satellite in a constant position with reference to the sun, thus providing consistent illumination throughout the year. The circular orbit permits uniform data acquisition by the satellite, and efficient command control of the satellite by the command and data acquisition (CDA) stations located near Fairbanks, Alaska, and Wallops Island, Virginia.

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\*The ITOS spacecraft will be launched into nominal 3 p.m. or 9 p.m. ascending node orbits.



## **2. Space Shuttle Characteristics**

Although the primary mission of the space shuttle is logistics support of the space station/base, alternate missions of satellite placement and retrieval, and satellite servicing and maintenance have been identified as being of major interest in future space program planning.

This study has been concentrated on examining the potential spacecraft program cost impact resulting from shuttle availability with the following advantages:

- Large weight and volume capacity
- Low g environment
- Capability of Spacecraft retrieval for repair, refurbishment, sensor updating and replacement
- Intact abort

The 15-foot diameter x 60-foot long cargo bay of the shuttle and the high payload capacity (50,000 pound maximum, for a 270-n. mile orbit with an inclination of 55 degrees) remove the prior spacecraft configuration restrictions relative to shape, size, and weight. Elaborate "black-box" packaging optimization requirements can also be relaxed.

The low-g environment with regard to launch and landing steady-state accelerations (maximum of 3 g) and the capability of utilizing vibration isolators to reduce the sine and random vibration load levels will permit significant cost reductions in equipment design and testing.

The capability offered by the shuttle to retrieve satellites for repair, modification or reuse invites standardization of basic building-block modular components for spacecraft. This modularization of equipment will make on-orbit satellite repair a feasible future procedure.

To minimize spacecraft losses due to launch vehicle failures, the space shuttle will be designed for intact abort. That is, if the shuttle must abort a mission, it will not be required to jettison the payload. The shuttle will be able to return to earth with its cargo intact.

### **B. KEY AREAS EXAMINED**

The study program was divided into three main categories: shuttle impact on general spacecraft configuration, shuttle effect on equipment and testing costs, and shuttle impact on overall future ITOS operational program costs.

As part of the ITOS/shuttle compatibility study effort, required changes to the existing ITOS satellite and new design approaches for future ITOS spacecraft were examined. Topics of exploration included shuttle interfaces, rendezvous and docking, emergency attitude control, and spacecraft deployment from the shuttle cargo bay.

Each spacecraft subsystem was examined for possible hardware redesign and cost reductions due to greater volume and weight allowables and a more benign environment. Manufacturing, integration and testing were evaluated for potential savings. Component reliability and techniques for extending "wear-out" life (radiation shielding, etc.) were explored in an attempt to make full effective use of shuttle capabilities in lowering equipment costs.

Cost models for a 1978 ITOS program, with and without shuttle availability, were generated to allow cost comparisons. A brief explanation of failure modes, failure rates, and "wear-out" has been included in this report as background for the systems reliability evaluations. The report concludes with a summary of the total potential cost reductions as a result of equipment design, testing, and program operation savings.

### **C. STUDY ASSUMPTIONS, CONSTRAINTS, AND EXCLUSIONS**

Basic parameters around which the study was formulated, and major constraints and exclusions resulting from internal assessment and NASA direction are presented:

- The Space Shuttle/Tug combination can deliver and retrieve spacecraft from any desired orbit.
- The shuttle benign environment will provide lift-off and landing accelerations of no more than 3 g's.
- The shuttle mission duration is approximately two weeks.
- The shuttle cargo bay is unpressurized and has a clear volume of 15 feet diameter x 60 feet long.
- The maximum shuttle payload capacity is 50,000 pounds (for a 270-n. mile orbit with an inclination of 55 degrees).
- The spacecraft program philosophy is based on a high probability of no outage.
- No launch or retrieval costs are included in the program cost models.
- All spacecraft repairs, component update, etc. will be done on the ground. In-space repairs and impact of extra-vehicular activity (EVA) have been excluded from system modeling because of the difficulty of establishing estimative manrating requirements.

- It has been assumed that spacecraft design and project support costs for the 1978 ITOS Program would be identical with or without shuttle availability.
- No launch failures are included in system modeling.
- All cost reduction evaluations are based on potential changes to the existing ITOS spacecraft design and testing. No cost analyses have been conducted on the completely new configurations described in Section IIB.

More detailed considerations and assumptions will be found preceding specific report sections, as applicable.

## SECTION II

### ITOS/SHUTTLE COMPATIBILITY

#### A. DESIGN CHANGES TO EXISTING ITOS SPACECRAFT

##### 1. General

Prior to a discussion of present capability and/or redesign requirements as related to shuttle compatibility, a brief description of the existing ITOS spacecraft is included.

The ITOS spacecraft is a rectangular box approximately 40 x 40 x 48.5 inches long. It has been specifically constructed to withstand the anticipated launch-vibration environment and to allow for future growth with a minimum of design changes. Three slides of the structure (a baseplate and two equipment panels) support the sensors and electronic equipment. The equipment panels are hinged at the baseplate and can be swung down to form a flat work area for unencumbered installation and testing of components and subsystems. During reassembly, the equipment panels are raised and joined to the adjacent sides and top structure to form the spacecraft. A three panel solar array, each paddle 36.4 inches wide by 65.2 inches long, is hinged to the top edges of the main structure. The panels are folded against the spacecraft sides in the launch configuration and are deployed into a plane normal to the sides by actuator devices after the satellite has achieved orbit. The end of the spacecraft continuously exposed to the sun supports a passive-control thermal fence, consisting of two concentric cylindrical fins. Active thermal-control louvers on the exterior of the equipment panels operate with the thermal fence to maintain spacecraft heat balance. A separation ring attached to the baseplate mates the satellite to the Delta N-6 launch vehicle.

The ITOS dynamics and attitude-control subsystem orients and stabilizes the satellite with respect to the earth and orbit plane. TV cameras and scanning radiometers remain earth-facing continuously as the satellite travels along its orbit. A momentum wheel assembly, the heart of the dynamics and attitude control subsystem, together with other pitch-control loop hardware and momentum coils serves to control the satellite's pitch attitude. The spacecraft's roll-yaw attitude is maintained by a quarter orbit magnetic attitude-control (QOMAC) coil, a magnetic bias-control (MBC) coil, and a pair of liquid nutation dampers.

The power supply for ITOS consists of a solar array of approximately 10,260 n-on-p 2x2-cm silicon solar cells, plus two redundant sets of shunt limiters, charge controllers, rechargeable nickel-cadmium batteries, voltage regulators, and protective devices. The power supply design includes fuses to isolate parts of the subsystem whose failure might cause total power loss.

The CDA stations control the operation of the satellite by programmed commands transmitted to the spacecraft. The command, control, and communication subsystems are completely redundant and permit tracking of the satellite at all times during launch and operation.

## 2. Shuttle Interfaces

The ITOS spacecraft structural, mechanical and electrical systems that are operational during initial lift-off and/or during the satellite pre-launch and launch phases must be compatible with the shuttle/tug.

As the cargo bay of the shuttle may have longitudinal rails for payload containment similar to those found in aircraft, it appears beneficial to support the spacecraft in the launch configuration at four points as shown in Figure II-1. These support points, fitted with vibration isolators to attenuate the sine and random vibration g levels induced by the shuttle, offer the advantages of better load path and reduced transmitted forces as compared to the present separation ring support arrangement. Geometric compatibility between spacecraft support spacing and rail spacing can be assured by the use of cross-beams on the rails as required.

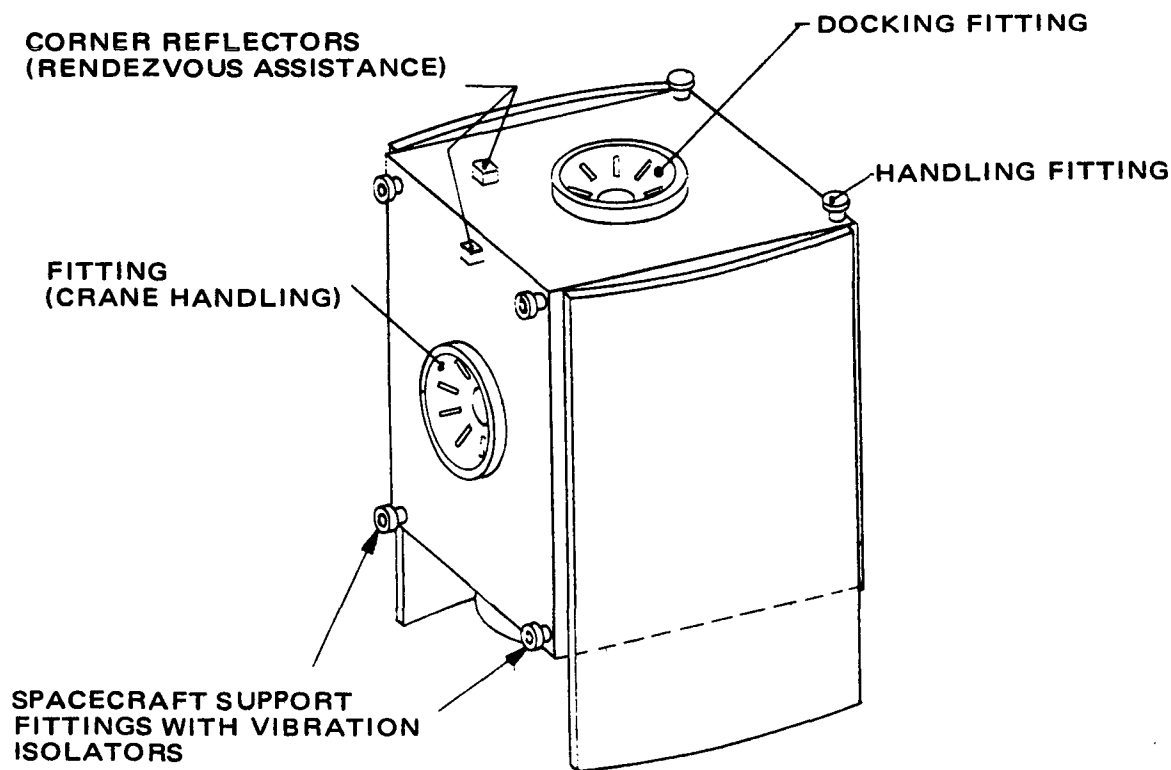


Figure II-1. Modifications To Existing ITOS Design

Although shuttle payload deployment schemes are topics of future studies, remotely controlled cranes or articulated arms are presently considered to be the preferred arrangements. Crane mating fixtures or simple ears at the spacecraft corners, as shown in Figure II-1, will enable safe handling of the ITOS satellite during the pre-launch and launch operations. Structural loading induced by these near-zero g field operations are very small, being only dependent on low acceleration forces.

It has been stated that the shuttle may be in orbit for longer than a week before complete payload deployment. To preclude battery drainage during this interval, it will be necessary to draw spacecraft standby-mode power from the shuttle electrical system. This should create no problems as the power demand is small and the connections can be easily disengaged by a minimum of inter-vehicular activity (IVA).

The degree of spacecraft protection required with shuttle operation has not been ascertained. As a minimum, a dust cover could be provided for use during shuttle loading and launch pad standby. This cover should provide adequate venting as the cargo bay is not pressurized.

### 3. Rendezvous and Docking

To enable retrieval from orbit, the existing ITOS spacecraft requires rendezvous enhancement devices and docking hardware. It has been assumed that the tug (retrieval vehicle) will be equipped with an illuminator and with search and docking sensors that can detect, track and range a "passively co-operating" satellite target equipped with cube corner reflectors (no active transponder). These highly reflecting optical corner cubes (from  $\frac{1}{4}$  inch to  $2\frac{1}{2}$  inches per side depending on sensor design) enhance the return of light energy back to the search vehicle. All attitude detection and tracking of the satellite requires approximately 20 of these retro-reflectors properly spaced on the spacecraft.

A mechanism to enable physical docking with the tug must be added to the satellite. It appears reasonable to assume the tug will have a docking probe that necessitates a compatible docking port on the spacecraft. An examination of the existing ITOS reveals the top of the spacecraft is the most unobstructed surface upon which to mount the docking equipment. The structure below the thermal fence can be readily reinforced to support docking and tug-induced injection and retrieval loads. This addition requires the relocation of the beacon antenna and a modification of the spacecraft thermal control system. Preliminary evaluations have indicated that minor resizing of the Active Thermal Controller will compensate for the reduction in passive thermal control capability. The spacecraft docking port is visualized as a conical device located so that its axis is colinear with the spacecraft pitch axis. An electrical contactor incorporated into the docking port would enable the spacecraft to draw standby power from the tug, if required.

#### 4. Emergency Attitude Control

A prerequisite to a docking maneuver is that the spacecraft maintain a stable attitude. Maximum tolerable spin rates allowable for coupling are in the order of one revolution per hour. Docking with the present ITOS with an operational attitude control system would present no difficulties as the spacecraft rotates about its pitch axis at a rate of one revolution per orbit (approximately  $\frac{1}{2}$  rph). However, a failure of the momentum wheel assembly resulting in wheel stoppage would create an unfavorable dynamics situation, as the spacecraft would spin at approximately 3 rpm about a maximum momentum axis which is rotated 70 degrees from the original pitch axis.

Methods of maintaining spacecraft stability with MWA malfunctions have been examined. One scheme would be to fold down the solar panel on the anti-earth side of the vehicle to keep the maximum momentum axis along the pitch axis. Although the spacecraft would eventually spin-up to 3 rpm, magnetic torquing, or, if time is of the essence, deployable yo-yo weights could be utilized to reduce the spin rate to the allowable docking limit.

Another approach would involve the utilization of a small, high speed auxiliary wheel functioning as an emergency gyro-stabilizer. Upon any spacecraft power failure or wheel malfunction situation, relays would activate this emergency wheel, which would be powered by its own reserve battery. The requirements for the capacity of this battery would depend on the time between failure detection and arrival of the retrieval vehicle.

#### 5. Component Accessibility and Ease of Replacement

The ITOS spacecraft presently incorporates desirable features as related to ease of component replacement. Entrance into an assembled unit is accomplished through openings in the two side panels. Should greater accessibility be required, the equipment panels may be easily opened outward to provide a flat work area. All equipment panels and solar array paddles are interchangeable with similar units. All boxes can be interchanged with like components with the use of simple tools. Because of the accuracy with which the structure is assembled and held together, replacement of cameras and radiometers can be accomplished with a minimum of re-alignment. The electrical harness and RF cables are located so as not to impede the removal of electronic equipment.

To facilitate possible future IVA/EVA\* repairs, captive hardware and quick disconnect fasteners would be used as required.

---

\*IVA Inter-Vehicular Activity  
EVA Extra-Vehicular Activity

## **6. Deployment and Launch of Spacecraft from Shuttle Cargo Bay**

The deployment and launch of an ITOS spacecraft from the shuttle cargo bay is depicted sequentially in Figures II-2 through II-4 and described later in the text.

It has been postulated that a space tug, carried aboard the shuttle, will be utilized to transport the spacecraft from the shuttle orbit to the satellite orbit. Another supposition, based on concurrent shuttle studies, is that a remotely controlled payload handling crane will be provided for in-space unloading and loading of tug and spacecraft. As little information is available relative to the tug and payload handling crane, the following general configurations and characteristics have been hypothesized for the purposes of this study.

The space tug, using a space storable bi-propellant propulsion system, is sized to execute a 12-degree plane change and a 530-n. mile altitude change from an initial 270-n. mile circular orbit. The payload capacity (in either the injection or retrieval mode) for these orbital transfers is 1000 pounds. The dry weight of the vehicle, which can hold two men, is 3000 pounds and the propellant load (including oxidizer) is 7500 pounds.

Two possible tug versions are shown in Figures II-5 and II-6. Both models have a pressurized compartment of spherical shape which houses the crew and all the controls and equipment necessary to man the tug. The crew transfer tunnel allows the astronauts to enter the vehicle from the shuttle crew compartment. The payload docking mechanism, well visible by the tug crew, is equipped with a locking device to couple with the spacecraft. A fitting for shuttle crane handling, similar to the device employed on the spacecraft, is provided for unloading and loading the tug into the cargo bay. The tug is equipped with the necessary communications, guidance and rendezvous systems. A major difference in the two tug versions is that the tug depicted in Figure II-6 is equipped with manipulatable arms to assist in the docking operation.

The payload handling crane is a remotely controlled articulated tubular boom having its base attached to a shuttle structural member located at the mid-length upper surface of the cargo bay.

The free end of the crane is equipped with a coupling head which mates with the crane fittings on the tug and spacecraft. The articulated joints of the crane and coupling head have the necessary freedom of controlled motion to assure safe coupling and handling of the payload in almost any position. The crane is operated remotely by the crew of the shuttle with direct visual and television monitoring.

Figure II-2 depicts the ITOS spacecraft and the space tug secured in the cargo bay of the shuttle. The satellite launch operation begins with the opening of the cargo bay door and the subsequent unloading of the tug by utilization of the



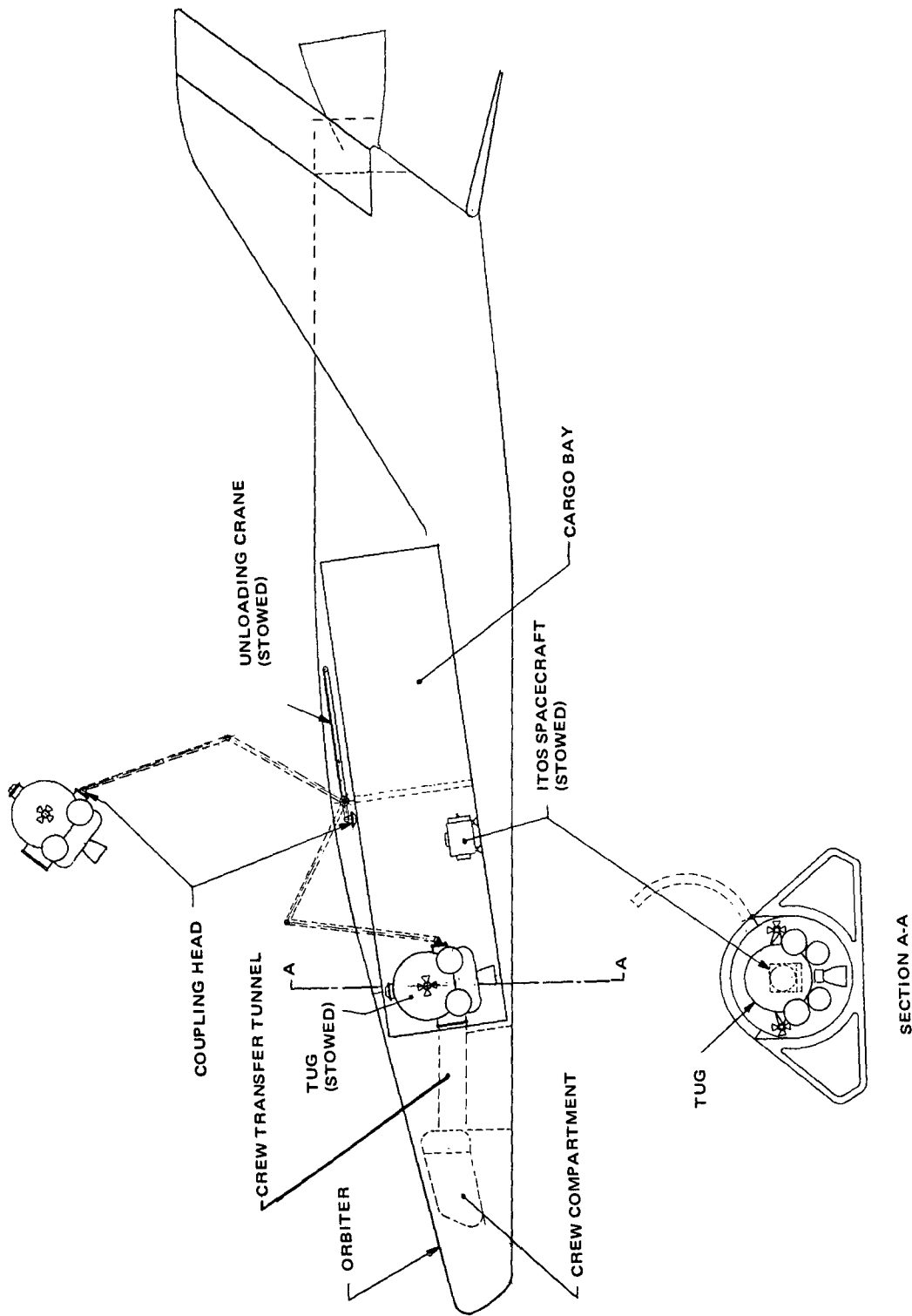


Figure II-2. Spacecraft/Tug Launching - Sequence 1

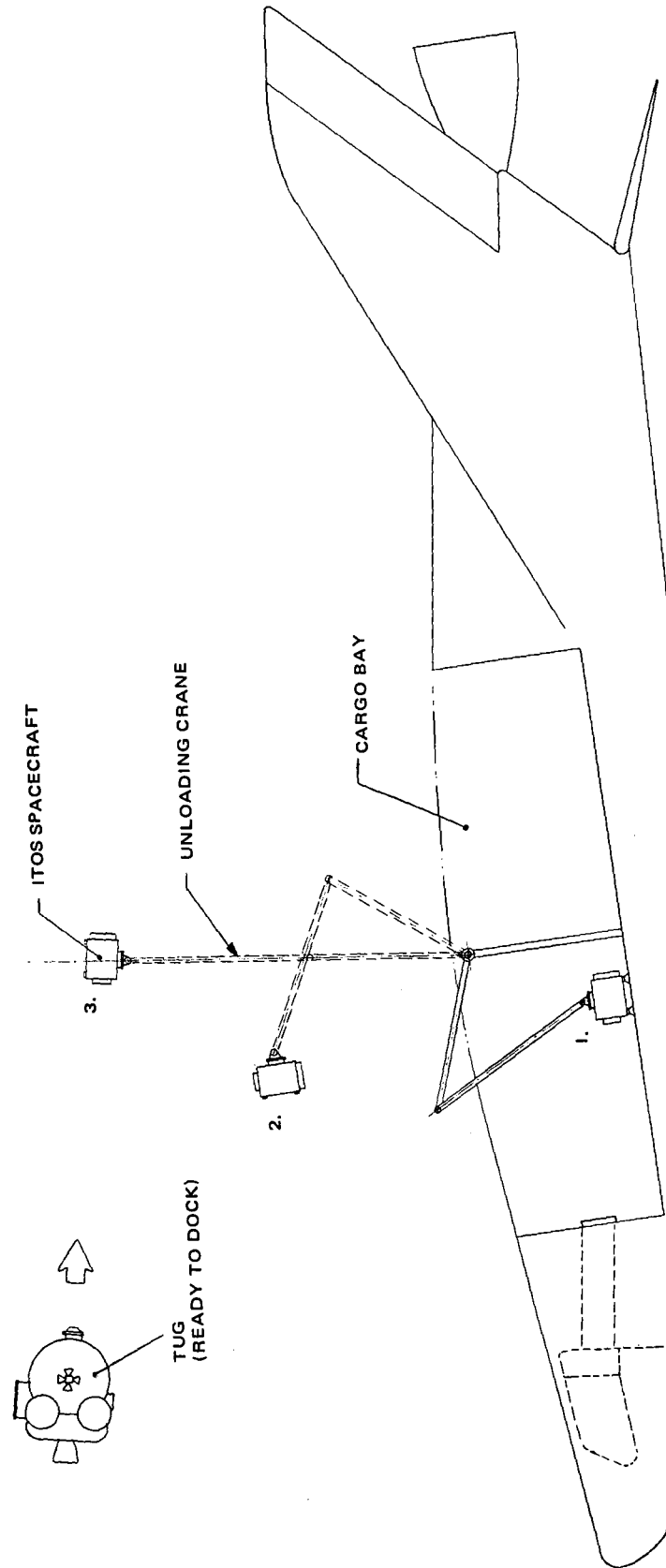


Figure II-3. Spacecraft/Tug Launching - Sequence 2

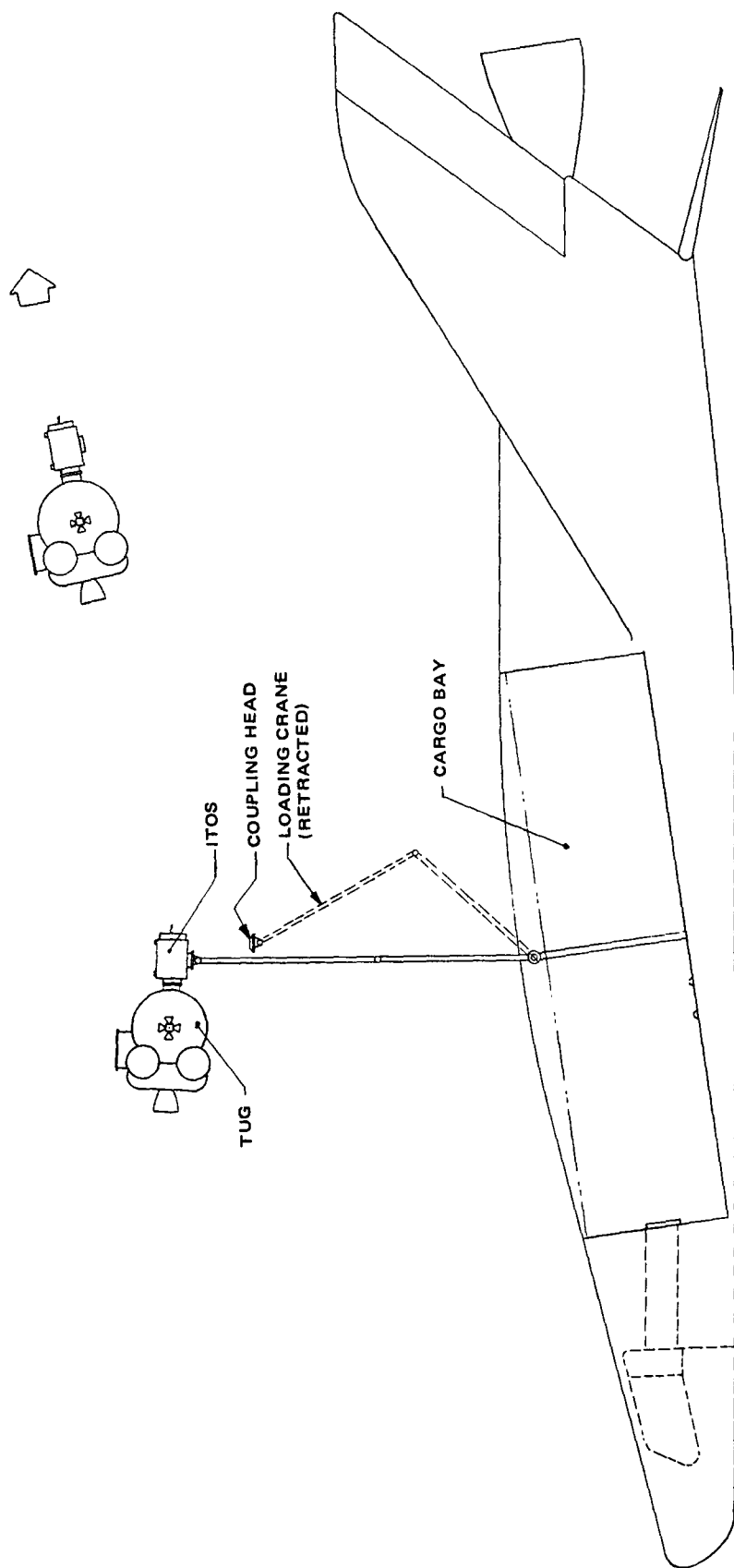


Figure II-4. Spacecraft/Tug Launching - Sequence 3

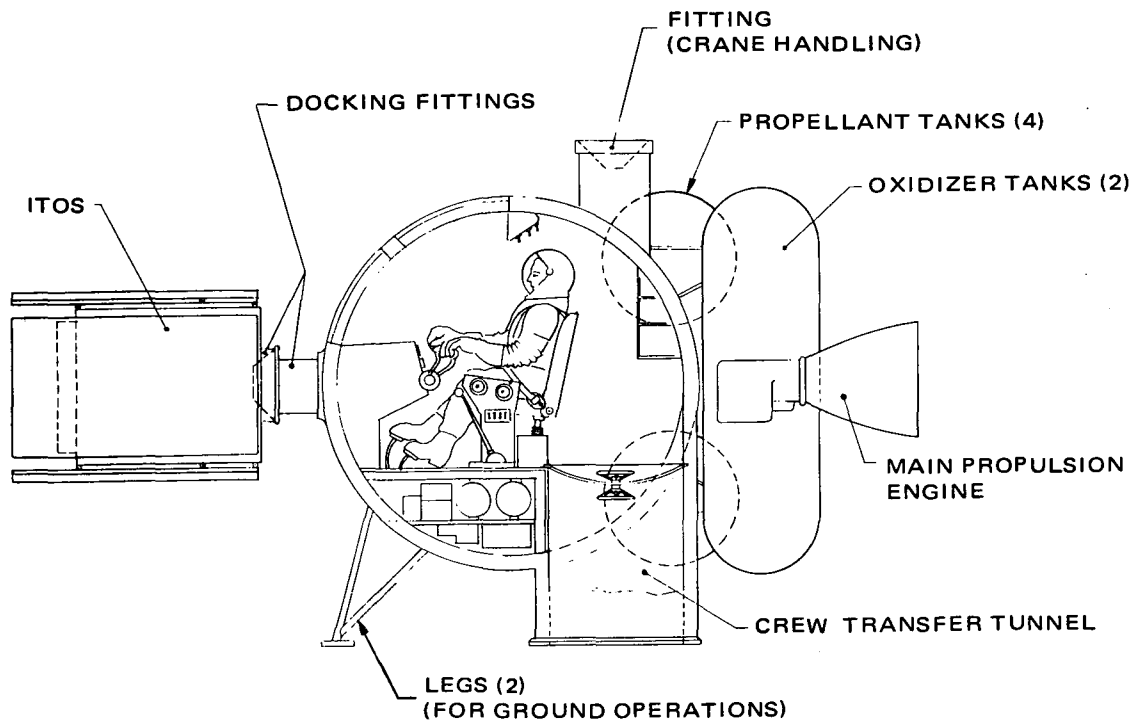


Figure II-5. ITOS/Tug – Docked Configuration

payload handling crane. Crane and tug separation occurs when the tug is clear of the cargo bay door and under full command of the crew.

Figure II-3 shows the tug in a position ready for satellite docking. Prior to deployment, a modest amount of IVA is required to disengage the spacecraft from the shuttle support rails, disconnect the standby-mode power line, and remove the satellite's protective covering. The unloading crane is then employed to transfer the spacecraft out of the cargo bay and into an extended boom docking position.

With careful use of its attitude control thrusters, the tug then docks or couples to the spacecraft as illustrated in Figure II-4. Upon completion of docking, the crane coupling head is separated from the satellite and the boom retracted.

After the handling crane has been secured, and the cargo bay door closed, the tug's main propulsion engine is ignited to deliver the spacecraft into orbit.

A retrieval mode loading of tug and spacecraft into the shuttle cargo bay will follow a reversed procedure.

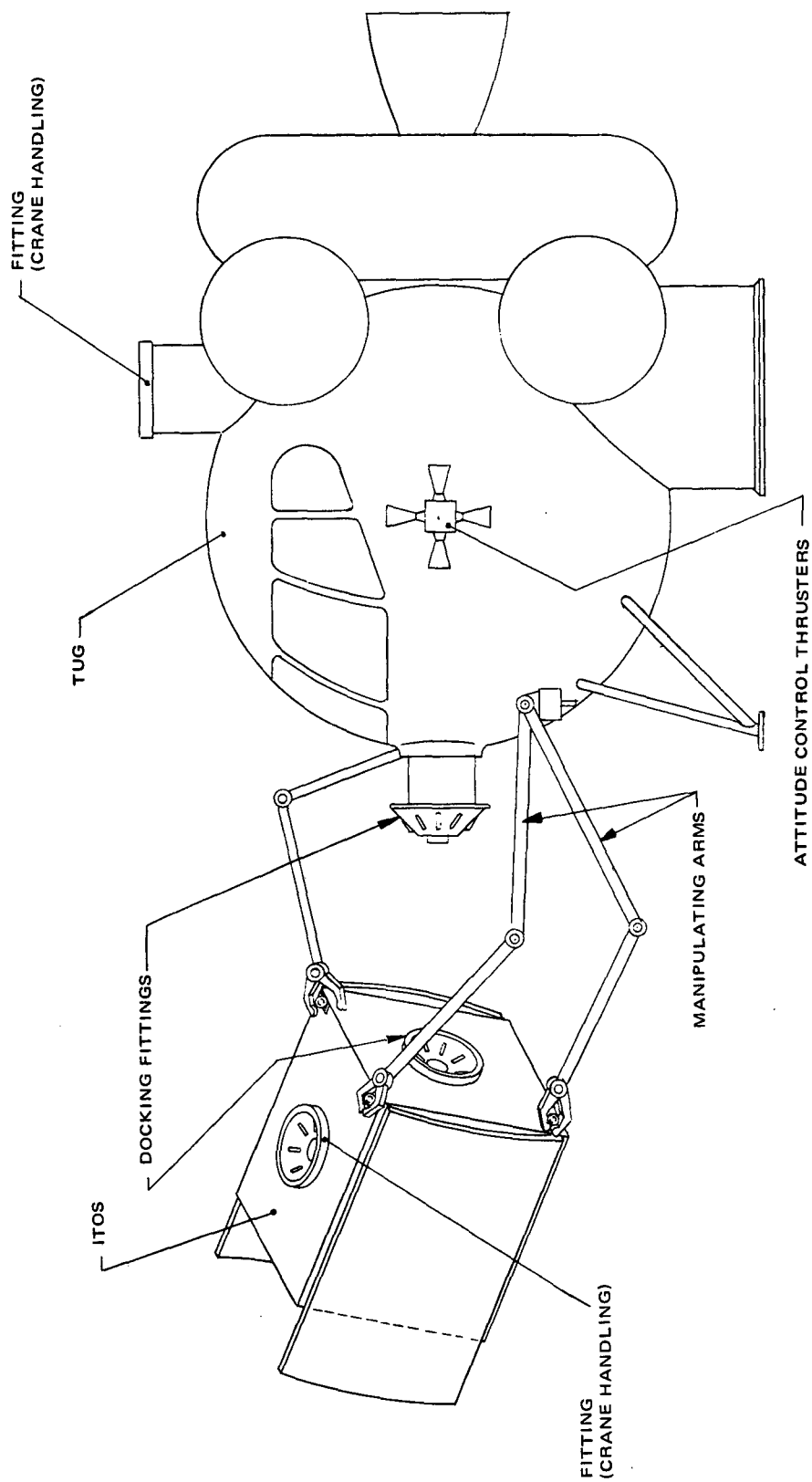


Figure II-6. ITOS/Tug Docking (Tug With Manipulatable Arms)

## B. NEW ITOS DESIGN APPROACH

The present ITOS has been designed to comply with the constraints of the Delta launch vehicle. However, completely new ITOS configurations based on shuttle availability will not be influenced by these restrictions and can take full advantage of the larger weight and volume allowables and the more benign launch environment. Such future ITOS designs would incorporate the features for shuttle compatibility as defined in paragraphs A2 through A6 of this section.

To gain more benefits from the repair, refurbishment and future EVA possibilities offered by the Shuttle/Tug, better equipment accessibility will be required. An "inside-out" configuration in which the main support structure is located internally (or centrally) and the equipment mounted externally as shown in Figure II-7, would facilitate replacement of faulty components by an astronaut. Heavier, thicker walled, box covers would provide more effective radiation shielding for the equipment. A stiffened thermal blanket attached to the support structure by means of quick-disconnect fasteners could be used to enclose the spacecraft and provide the required thermal environment.

Another structural concept which will provide ready accessibility to the spacecraft equipment is the modularized configuration depicted in Figure II-8. This concept permits the replacement of an entire defective module in a short time.

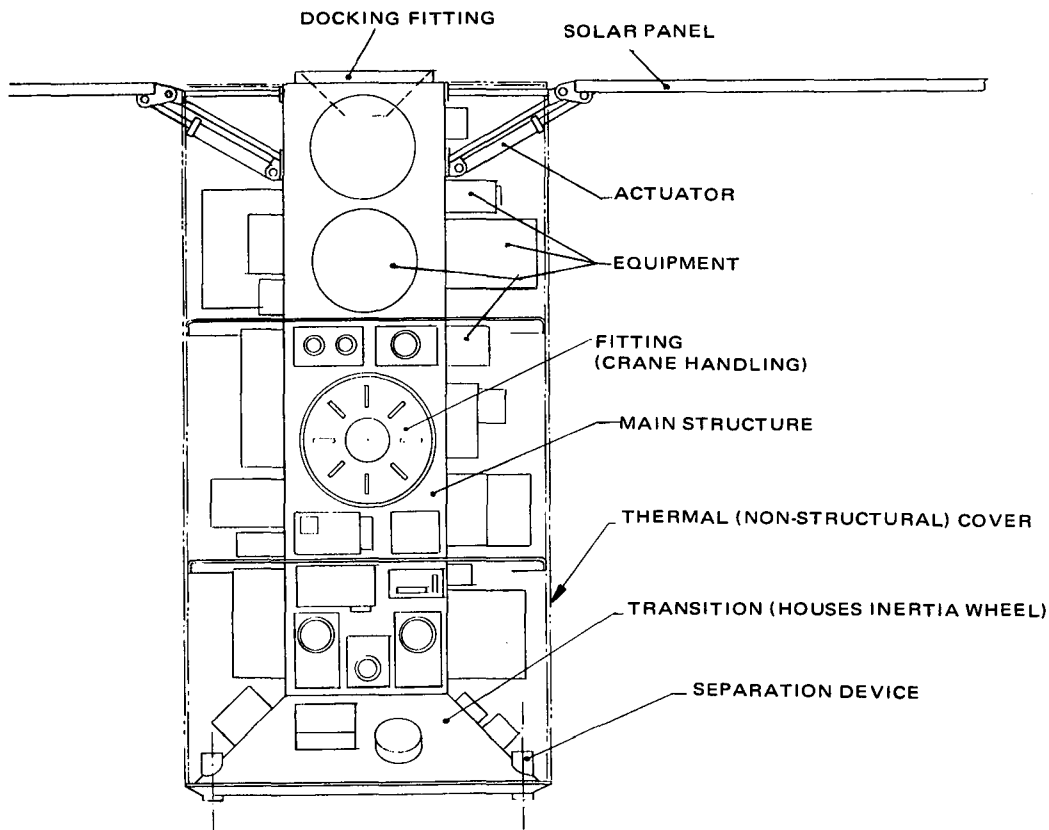


Figure II-7. "Inside-Out" Spacecraft Concept

The modules may contain parts of, or a whole, subsystem. Outside test plugs may be provided to determine which unit contains a defective component. This spacecraft does not require a thermal blanket as the individual modules are equipped with their own thermal control. An intermodular harness and connector system automatically provides electrical continuity between adjacent modules upon "stack-up" and mechanical coupling of the units.

These illustrative structural arrangements are but a few examples of possible future ITOS spacecraft configurations. The number of feasible concepts is only limited by the imagination of the designer.

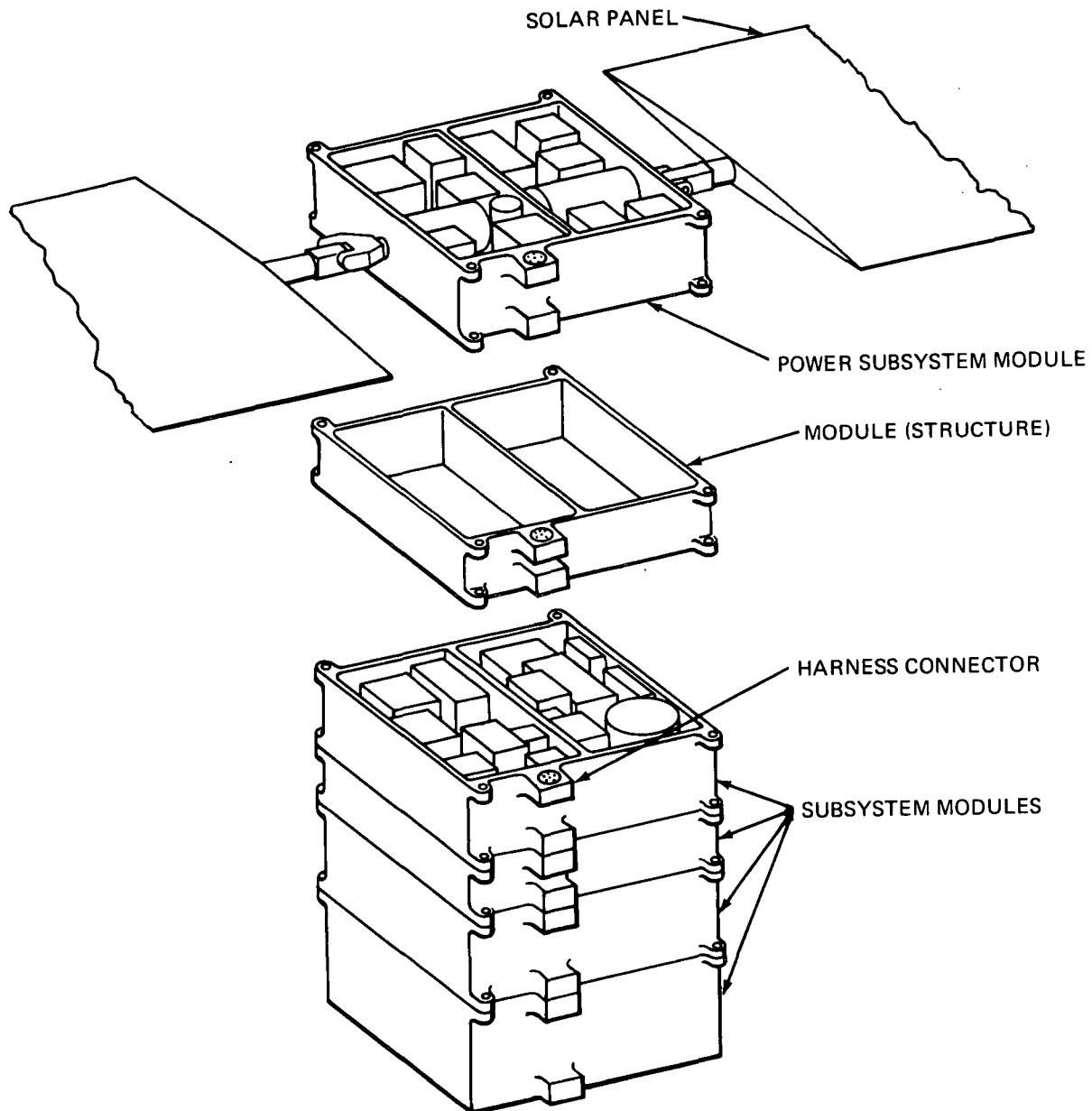


Figure II-8. Modularized Spacecraft Concept

## SECTION III

### EQUIPMENT IMPACT

#### A. GENERAL

For each subsystem, possible hardware redesign and cost reductions due to greater volume and weight allowances and a more benign launch environment have been evaluated. The impact on manufacturing and component testing has been studied and a new testing philosophy proposed.

The ITOS subsystems examined for potential cost savings are:

- Power Supply
- Sensors and Experiments
- Command and Control
- Structure
- Thermal
- Communications
- Dynamics

#### B. POWER SUPPLY

The power supply subsystem consists of a solar array, batteries with tapered charger, a shunt limiter, an unregulated bus and a series regulator. This subsystem has been reviewed for shuttle-induced cost savings and the following cost effective redesigns have been determined:

- Use of a flat array panel. The present panel is curved to meet the shape and volume constraints imposed by the Delta launch vehicle. If these constraints are removed, the panels could be fabricated flat, and the manufacturing cost reduced, although there would be minimal initial tooling costs.
- Use of thicker cover glass to reduce array degradation due to particle radiation. With thicker cover glass the array area required, and hence costs, are decreased since array sizing is a function of end-of-life power capabilities.
- Elimination of honeycomb construction in the panels. The present array is mounted onto a honeycomb panel to minimize weight. If the solar cells were mounted on a stiffened sheet metal panel, the weight would increase but fabrication cost would decrease.



### C. SENSORS AND EXPERIMENTS

The primary sensing-experiment functions of the present ITOS-1 spacecraft are performed by the Scanning Radiometers with associated processor and tape recorders, the Vidicon Cameras with associated electronics and tape recorders, and the Automatic Picture Transmission units with associated electronics. Secondary sensors consist of a Flat Plate Radiometer and a Solar Proton Monitor with associated converters and tape recorder.\*

Small savings in material and fabrication costs are possible by replacing all magnesium housing material with equivalent strength aluminum alloy. Despite this minor saving, the review has shown that material and fabrication costs are for all practical purposes independent of the launch mode. Mission performance and power requirements in the space environment have governed the design. Even where miniaturization resulted, superior performance capability was the major factor affecting trade-off selection from alternatives.

The major cost reduction from the use of the shuttle arises not from hardware savings but from a change in test philosophy which decreases or eliminates environmental acceptance testing of components as explained in paragraph I of this section.

### D. COMMAND AND CONTROL

The command subsystem is comprised of the following units: decoder, programmer, command distribution unit, time base generator and time code generator. Minor savings in material and fabrication cost are possible by replacing the magnesium in these units with equivalent strength aluminum and deleting the weight-saving final machining operation.

Again, material and fabrication costs are for all practical purposes independent of the launch mode, with performance and power requirements governing the design. Consequently, there are virtually no shuttle-induced cost savings from design, material or fabrication changes. As with the power system, the major cost saving from the use of the shuttle results from a change in test philosophy.

### E. STRUCTURE

The ITOS spacecraft structural configuration is comprised of a main body with three deployable solar panels. The main body consists of externally reinforced aluminum panels bolted together to form a rectangular box. Sensors

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\*The next generation ITOS spacecraft will most probably not have television sensor systems.

and "housekeeping" equipment are located on the two equipment panels and the baseplate. The two equipment panels are hinged to the baseplate permitting them to be folded open to facilitate assembly, test and checkout of components.

The equipment panels and baseplate are of orthotropic stiffened plate construction with reinforcing gussets as required. This structural configuration was selected over all other alternatives on the basis of:

- Efficient area utilization and structural loading.
- Compactness for efficient thermal environmental protection.
- Simplified integration and checkout, and better accessibility.

A reduction in launch loads or the elimination of weight and volume constraints as is predicted for the shuttle launch, should contribute very little structural cost savings. The structure is already of simple stringer and skin construction so that any reduction in load capability (if warranted) would have a negligible cost impact. Also, the original specification volume constraints did not greatly influence structure cost, and the lifting of these constraints would not represent a cost saving.

As is evident, many other factors contributed to the final ITOS structural design rather than low weight per se. However, small cost savings can be realized in the area of parts standardization and fabrication simplification, and include:

- Deletion of all lightening holes.
- Use of constant thickness aluminum skin and constant size channels. (Savings accrue from part standardization.)

#### **F. THERMAL CONTROL SUBSYSTEM**

The ITOS thermal design incorporates a passive system, augmented, as required, by an active thermal control system. The principal elements of the passive system are the thermal fence compensating solar absorption device, fixed radiating areas, and multilayer insulation. The active control system consists of four independent, hydraulically actuated louvers (two per equipment panel), which provide variable emissivity radiating areas as a function of spacecraft temperature level.

The thermal control system costs are rather insensitive to weight, volume and benign environment considerations and consequently there is little cost saving attributable to shuttle availability except in the component testing area.

## **G. COMMUNICATIONS**

The communications system consists of four separate RF links, each consisting of an electronics grouping, transmitters, and an antenna subsystem. The only appreciable cost savings is in the antenna subsystem area.

The real-time antennas are deployed after orbit injection because of booster fairing envelope constraints. Material and fabrication costs can be decreased by using fixed antennas. Further savings in this area can be achieved by eliminating one of the redundant real-time antenna pairs which was provided to improve the reliability of a deployable antenna subsystem. The inherent high reliability of a fixed antenna eliminates the need for antenna redundancy. The cost of a second array, notch filter, hybrid coupler, and termination and cable set, can be saved at the expense of adding a coaxial relay and of sustaining a modest splitting loss.

## **H. SPACECRAFT DYNAMICS**

The spacecraft dynamics control subsystem is configured to align the spacecraft pitch axis parallel to the orbit normal and to continuously orient the yaw (camera) axis along the local vertical. The subsystem comprises a Stabilite pitch axis control loop, magnetic roll and yaw axes controls, momentum coils, nutation dampers and related sensors.

Cost evaluation of this subsystem has shown potential cost savings in the areas described in paragraphs H1 to H3, which follow.

### **1. Pitch Control**

Cost savings in the momentum wheel assembly can be obtained by simplifying its design and manufacture at the expense of added weight. A titanium housing is currently used to avoid undesirable thermal gradients across the bearing, without sustaining a weight penalty. Cost savings can be achieved by substituting less costly, but heavier, materials, that achieve the same purpose.

### **2. Magnetic Coils**

The use of magnetic coils with larger diameter, heavier wire would eliminate breakage problems associated with coil winding and terminal connections, thus resulting in a cost saving.

### **3. Nutation Dampers**

Reduced costs could be realized by constructing the nutation dampers from heavier wall tubing, which is easier to fabricate; and by utilizing a heavier, less expensive expansion bellows.

#### **I. TEST PHILOSOPHY**

The current test philosophy for ITOS is governed by applicable NASA/GSFC specifications. Representatives of each component are normally subjected to component qualification test, then every flight component (including spares) must be subjected to a component acceptance test. The spacecraft is also subjected to a total system acceptance test for launch readiness.

The use of the shuttle in place of the present launch booster is expected to reduce launch loads, but more important, it will permit less costly replacement of orbiting spacecraft through retrieval and refurbishment.

The former (launch load reduction) has little effect on the costs of ITOS testing. While the launch vehicle and in some cases the apogee kick motors may determine the shape of the test vibration spectrum, the level tested is generally in excess of the actual sustained environment for "over-test" reliability purposes. Granting that a test vibration level decrease could be justified, there would be virtually no test cost saving except those obtained whenever test tolerances are loosened.

The latter (less costly replacement) could have a measurable effect on the cost of ITOS testing, providing the overall NASA/GSFC specifications are revised accordingly.

By virtue of the simplified refurbishment made possible by the shuttle, the overall test philosophy for ITOS could be changed to that accorded to manned aircraft. This philosophy requires that representatives of each component be subjected to a set of Qualification Environment Tests. Levels are stringent so that ordinarily a separate component acceptance test would not be required for prime units. A Qualification Test Review would determine those components exhibiting special problems and only these would require component acceptance tests. Spacecraft acceptance testing would remain unchanged.

## **SECTION IV**

### **SYSTEM AND OPERATIONAL IMPACT**

#### **A. SPACECRAFT PROGRAM**

##### **1. Assumptions**

The 1978 ITOS spacecraft program upon which system and operational cost impact of shuttle availability will be based is dependent on the following assumptions.

- Present ITOS mission requirements will be in effect.
- 10-year operation with minimum outage (high probability of no outage).
- Complete sensor update every two years.
- Design freeze on basic spacecraft configuration and "house-keeping" equipment.
- Sensor updating requires no spacecraft redesign.

##### **2. Baseline System**

The baseline system (no shuttle availability) has been defined as a 10-year program requiring complete sensor updating every two years. The program requires a total of ten spacecraft, two of each model. Redundancy exists in all subsystems.

At the beginning of each 2-year cycle, a spacecraft with the latest type sensors will be launched. The second spacecraft of that specific model would be stored and serve as a back-up unit ready for immediate injection. In case of an inflight system failure, the back-up spacecraft would be launched into orbit and remain in standby mode. Complete activation of the second satellite and total turn-off of the first would occur upon failure of an entire redundant system.

#### **B. RELIABILITY, REPAIR, AND REPLACEMENT PHILOSOPHY**

##### **1. General**

In order to understand the relative system reliabilities with and without shuttle operation, the following background information is presented.

## 2. Failure Mechanisms

There are three failure mechanisms to which all failures can be attributed; namely, wearout, infantile failure, and random failure.

### a. WEAROUT

Wearout is the failure mechanism most closely related to human experience and therefore the one most easily understood. Fatigue, a wearout conception is something which all people feel; it approximates the true nature of the wearout process, i. e., becoming useless as the result of use. Wearout is a very predictable failure event which is "normally" (Gaussian) distributed about some "Mean Time To Wearout." For this reason it is required that the mission lives of all components be sufficiently shorter than wearout life so as to minimize the probability of failures by this mechanism.

### b. INFANTILE FAILURE

Infantile failure is also known as burn-in or early failure. A percentage of a part population is found to be weak or substandard, usually as a result of workmanship, and will fail very early in life. These infantile failures are hopefully weeded out during extensive preflight testing.

### c. RANDOM FAILURE

The random, or chance, failure is the most conceptually difficult to appreciate. It is generally thought of as failure resulting from the unfortunate simultaneous combination of:

- Low part strength (yet within specification limits) or low strength in a part parameter or characteristic which can't be measured or tested directly, and
- A higher than expected stress which is infrequent and usually the result of system complexities which makes its anticipation and prediction by the design engineer extremely difficult if not impossible.

In order to avoid random failure, two steps are taken in the design of a spacecraft:

- The part strength margin is effectively increased by utilizing a derating policy, and

- A worst-case analysis is performed to try to determine what maximum stress will be experienced under the worst anticipated conditions.

As a result of these policies, extremely low failure rates have been established. The soundness of such selection has been verified by inflight data over a period of years. However, despite redundancy which exists in many spacecraft subsystems, when the many thousands of parts in a spacecraft are factored into an overall reliability for the entire system, the probability of a failure is not negligible, especially for longer mission periods now being investigated.

### 3. Effect of Failure Theory on Interchangeability of Parts or Systems

Infantile failures are eliminated by testing and wearout failures by proper design; hence, a reliability prediction is basically interested in random or chance failures only. There are certain characteristics to this random or chance phenomenon, the major one being that the failure rate is constant. This implies the following:

- (1) The probability of success  $P_S$  (i. e., not having a failure) for a period time is a function only of the failure rate and the time interval of interest.
- (2) The  $P_S$  for just the 1st hour of operation is the same as for just the 10th or 1000th hour (assuming operation is still out of the wearout region). In the constant failure rate region the equipment does not care where it has been or how many hours it has seen. Only the interval ahead matters.
- (3) For one resistor the  $P_S$  is the same as for another similar resistor in their 1st, or 10th or 1000th hours. Also,  $P_S$  for one resistor for its 1st hour of life is the same as for the 1000th hour of life of another similar resistor. Similarly, if one spacecraft has already flown a year (and is not in the wearout region or had those components replaced which would be in that region during the next year), its probability of success in the 2nd year is the same as a completely new spacecraft for its 1st year. If the failure rate and interval are the same, the  $P_S$  is also the same. In other words, when looking at an upcoming interval of time in the chance failure region (constant failure rate), a new spacecraft is no better nor worse than a similar unit that has had previous space usage.

## **C. ITOS/SHUTTLE SYSTEM**

### **1. Reliability Considerations**

The ITOS/Shuttle System is based on the following reliability considerations:

- (1) No launch or retrieval failures.
- (2) When a back-up spacecraft is launched while the primary satellite is still functioning (but on its redundant system as a result of a failure), the newly launched satellite will remain in a standby mode and not be fully powered until it is needed.
- (3) Any spacecraft about to be launched will not reach its wearout life during the period of interest, i. e., the next cycle.

Item (3) will be achieved through wearout lives of sufficient length or by replacing components whose operating times may be approaching wearout life. Since electronic parts have lives on the order of 10 years, and since the lifting of weight restrictions on a shuttle system will allow increased shielding to minimize radiation degradation for 10 years or longer, wearout will not be a problem in the permanent housekeeping equipment. These subsystems are mainly electronic and will see only an average of 5 years of operational life in the 10-year period. The sensor systems, which contain most of the parts with lower wearout life, will be updated every 2 years, and with adequate design, wearout is expected to be no problem.

### **2. ITOS/Shuttle System Alternates**

#### **a. SYSTEM PHILOSOPHY**

With the shuttle system and its capability of satellite retrieval, the possibility of spacecraft reuse can be considered. Since only the basic spacecraft structure together with electronic housekeeping components with long wearout lives will be reused, and the sensors with shorter wearout lives will be replaced, this plan is possible from a wearout point of view.

Cost saving without sacrifice in reliability is the desired goal of the shuttle system, and the major question to be asked is, "What is the fewest number of satellites required in a shuttle system in order to achieve reliability equal to that of the baseline system?"



b. SHUTTLE SYSTEM NO. 1

The simplest shuttle system (No. 1) without outage would require only two satellites. One would be operational for the first two years; the second, for the next 2 years; then back to the first for the third two-year cycle, etc. Since the satellites are not near wearout, the reliability for a given period of time is the same for a new as for a used satellite, and the probability of success (reliability) of the baseline vs shuttle system No. 1 is given by:

Baseline System (No Shuttle)

$$P_{10} = \left[ e^{-\lambda_e t} (1 + \lambda_e t) \right]^5 \quad (IV-1)$$

Shuttle System No. 1

$$P_{10} = \left[ e^{-\lambda_e t} \right]^5 \quad (IV-2)$$

where

$t$  is the 2-year period

$\lambda_e$  is the effective failure rate of the satellites

$P_{10}$  is the reliability for 10 years

By inspection it can be seen that the baseline system, with its factor of  $(1 + \lambda_e t)$ , which will always be greater than "one", is inherently more reliable than Shuttle System No. 1.

c. SHUTTLE SYSTEM NO. 2

The next simplest shuttle system (No. 2) would incorporate three satellites, with redundant subsystems, operated in the following manner: (Refer to Figure IV-1)

- (1) At the beginning of each 2-year cycle, the primary spacecraft with the latest type sensors is launched.
- (2) The back-up spacecraft (same model as primary) is launched when an initial system failure occurs in the primary spacecraft, and remains in a standby mode.
- (3) The back-up spacecraft is completely activated upon failure of an entire subsystem in the primary spacecraft.

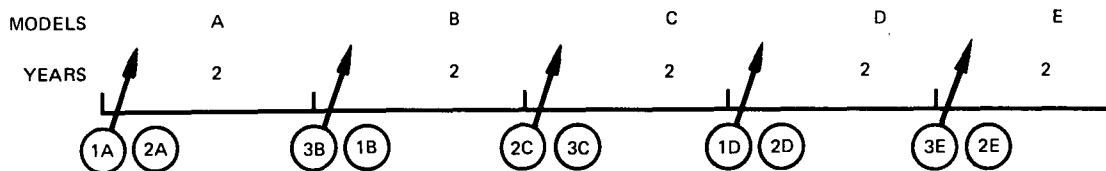


Figure IV-1. Program Model with Shuttle

- (4) At the end of a 2-year period, a third spacecraft, with updated sensors, is launched and the first two spacecraft are retrieved for retrofitting and sensor updating.

This cycle is repeated for the entire 10-year program life as depicted in Figure IV-1. Assuming that a back-up spacecraft will be in orbit an average of 1 year during every 2-year period, each spacecraft will have been operational in space for approximately 5 years at the end of the program.

The reliability of Shuttle System No. 2 is given by:

$$P_{10} = \left[ e^{-\lambda_e t} (1 + \lambda_e t) \right]^5$$

In analyzing this system, it is evident that by having a second back-up spacecraft, the same reliability is achieved as in the baseline (no shuttle) system. This shuttle system, with only three spacecraft, has the same reliability as the baseline system requiring ten satellites. Henceforth, Shuttle System No. 2 will be identified as the ITOS/Shuttle System.

## D. PROGRAM COST MODELS

### 1. Assumptions

- Launch and retrieval costs are not included.
- Cost comparisons for recurring costs are made against present ITOS program.

- Design and program management costs including program support, documentation, evaluation and analysis, and spares will be identical for the baseline and the shuttle system.
- ITOS costs will be constant over the 10-year period considered.

## 2. Cost Models

The common base for the system models is identical reliability, i. e., same probability of mission success. The advantage of a higher reliability would be difficult to trade off against a higher final program cost.

The cost models representing the 1978 spacecraft program without and with shuttle availability are as follows:

### The 1978 Baseline Program

- Defined in paragraph A2 of this section.
- Formulated from 1970 ITOS costs.
- Serves as primary comparative system.
- Requires 10 spacecraft.

### The 1978 ITOS/Shuttle Program

- Defined in paragraph C2 of this section.
- Employs shuttle for injection and retrieval.
- Requires three spacecraft and seven retrofittings (sensors, recorders, processors).

Also evaluated was a hybrid program with baseline-type operation (10 spacecraft) utilizing the shuttle for launch only. A final cost for this program would represent savings due to the equipment impact of the shuttle but not from its retrieval capability.

## SECTION V

### SUMMARY OF SPACE SHUTTLE EFFECT ON 1978 ITOS PROGRAM COSTS

#### A. COSTING ASSUMPTIONS

Cost analyses of the factors presented in the preceding sections reveal that savings can be realized in the 1978 ITOS program if a space shuttle is utilized. These economies can be seen in recurring spacecraft costs as well as in total program costs. In addition to the program assumptions of Section IV, paragraphs A and D, the following conditions and qualifications for cost comparisons were used:

- (1) Estimated savings are based on 1970 costs.
- (2) The 1978 state of the art is uncertain. This assumption was imperative because of the weight and volume trade-off. In most cases, miniature components have, or will become standard; therefore, no savings are indicated by the use of less compact equipment. Furthermore, many operations which are not common with commercial fabrication, assembly, and test, such as conformal coating and preconditioning, are now standard, space-oriented operations that are not considered to be cost tradeable.
- (3) No obvious savings are presented in design cost. Parameters of the Shuttle Study tend to indicate a design simplification over current thinking in such areas as radiation effects analysis because more shielding can be added; or designing an optimized and flexible bus, currently booster constrained. However, the shuttle launch and retrieval will create new and unique design problems resulting in a cost stand-off.
- (4) Since TOS/ITOS has accomplished 100-percent launch success, no savings were studied for initial mission accomplishment.
- (5) NASA specification relief has not been included as an assumption except in the areas involving weight, volume, and testing for infinite life. The weight and volume relief indicates that design verification testing from a part through the final spacecraft is required, but that savings in testing on a recurring basis, except for certain critical components, can be greatly reduced. This test philosophy can be passed along to vendors.
- (6) In maintaining a posture of high reliability and performance, workmanship standards, spacecraft reliability and performance requirements have not been relaxed. In fact, a future design optimization study to include more spacecraft redundancy is recommended.

- (7) Recurring costs in the program support areas have been reduced almost 40 percent from current levels prior to undertaking a 1978 cost model. These reductions result from repetitive spacecraft and are manifested in the following areas:

- Size of project management force
- Evaluation and analysis
- Documentation
- Spares philosophy and costs
- Integration and test
- Storage and exercise costs
- Rework, maintenance, and repair

#### B. EQUIPMENT COST IMPACT (SUBSYSTEM LEVEL)

Equipment cost saving techniques resulting from shuttle availability have been identified in Section III. These methods of cost reduction, which include hardware redesign and reduced component testing, have been factored into the most current ITOS cost budget to arrive at estimated quantitative cost savings for the spacecraft. Table V-1 depicts these equipment savings as a percentage of total subsystem costs.

TABLE V-1. EQUIPMENT COST IMPACT

Subsystem	Estimate of Potential Cost Savings
Power	25%
Sensors and Experiments	5%
Command and Control	5%
Structure	10%
Thermal	5%
Communications	6%
Spacecraft Dynamics	3%

The basis for the power subsystem saving and the associated computations are typical of the cost-saving derivations for all the subsystems. The power subsystem saving is primarily in the solar panels. The use of a flat sheet-metal type of construction for the solar panels instead of the present curved honeycomb construction in conjunction with reduced testing would yield a 50-percent saving in recurring fabrication and test costs. The fabrication and test cycle accounts for approximately 50 percent of the recurring costs for the panels.

Further, the use of thicker cover glass for the solar cells would minimize radiation degradation, thus permitting a 10 percent reduction in the required array area.

The power subsystem cost saving of 25 percent (Table V-1) was calculated as follows:

	<u>Normalized Cost of Present Power Subsystem</u>	
Solar Panels	0.70	
Power Supply Electronics	0.15	
Batteries	<u>0.15</u>	
Total Power Subsystem	1.00	
(1) Reduced panel cost due to 10-percent area reduction	0.70 - 0.07	= 0.630
(2) Reduced fabrication and test costs (50-percent reduction in fabrication and test costs, which represent 50 percent of panel cost)	0.25 x 0.630	= <u>0.158</u>
Total reduced normalized cost of panels, item (1) minus item (2)		0.472
Overall power subsystem saving due to reduced panel costs	0.70 - 0.47	= 0.23
Plus 2-percent cost saving in testing of power supply electronics unit and batteries		<u>0.02</u>
Total power subsystem saving		0.25 (25%)

### C. SPACECRAFT COST IMPACT

Assuming 1970 ITOS costs, reduced weight and volume constraints, and a benign environment, spacecraft cost savings were derived for the 1978 models and are presented in Table V-2. In these calculations a shuttle launch was assumed but no additional test savings due to retrieval test philosophy were included.

**TABLE V-2. SPACECRAFT COST SAVINGS**

Category	Cost Saving
Subsystem Hardware Savings (From Section V, paragraph B)	6%
Spacecraft Integration and Test Savings	10%
Total Spacecraft Savings	7%

A retrieval mode would require the addition of equipment on the spacecraft for rendezvous, docking and emergency attitude control, etc. Such equipment costs should be the subject of a future study. Obviously, they would tend to offset any saving realized by the reduced specifications.

#### **D. TOTAL PROGRAM COST IMPACT**

Final results of the cost modeling exercise, as summarized in Table V-3, are based on design and program support costs remaining constant for the ITOS baseline and ITOS/Shuttle programs. The ITOS baseline program assumes present specifications and launch requirements.

A modified ITOS baseline program with shuttle launch but no retrieval reflects only the equipment impact due to reduced specifications on the base-line system because of shuttle allowable payload and benign environment considerations.

The ITOS/Shuttle program with retrieval assumes a decreased number of total spacecraft, reduced test costs permitted by new retrieval testing philosophy, and the inclusion of IRAN (Inspect and Repair As Necessary). Equipment impact due to reduced environmental and size specifications has not been included because of the added requirements for unique retrieval hardware.

#### **E. RECOMMENDED FUTURE EFFORT**

The degree of comprehensiveness of this study has been established by the shuttle/tug information made available. Many questions, as previously presented by the cognizant NASA project directorate, must be answered prior to a more definitized evaluation of the shuttle impact on the ITOS program.

A key area of further exploration is the effect of EVA on program costs. The establishment of the attendant man-rating requirements may significantly affect program philosophy.

TABLE V-3. COST IMPACT OF SHUTTLE ON ITOS 10-YEAR PROGRAM

Program Cost Model	Spacecraft Program Cost (normalized) (See Note 1)	Total Program Cost (normalized) (See Note 2)
ITOS Baseline Program (No Shuttle)	1	1
ITOS Modified Baseline Program (Shuttle Launch Only)	0.93 (See Note 3)	0.96
ITOS/Shuttle Program (With Retrieval)	0.62 (See Note 4)	0.81
<p>Notes</p> <ol style="list-style-type: none"> <li>1. Includes cost of hardware fabrication, assembly, testing, flight acceptance testing, and spacecraft integration and testing.</li> <li>2. Includes design and program support costs.</li> <li>3. Reflects total spacecraft cost savings of 7% shown in Table V-2.</li> <li>4. Includes sensor updating and IRAN costs.</li> </ol>		

More extensive design effort will be required in the rendezvous, docking, and retrieval areas. Costs for implementing these operations have not been included in this study and could substantially alter the present conclusions.

The defining of a new spacecraft configuration, based exclusively on shuttle launch and retrieval considerations, should be included in the future work scope. This design should be in sufficient detail to enable a valid cost estimate.

Finally, more intensive investigations should be conducted in the areas of system safety requirements, payload handling, and deployment system interfaces.



## APPENDIX

The appendix contains work done during the early phases of the study in compliance with the initial NASA work statement. The four major topics examined were:

- A. MISSION ANALYSIS: The basic ITOS mission determines the ITOS orbit.
- B. SHUTTLE PERFORMANCE: The shuttle performance characteristics determine the payload capability for each orbit.
- C. TRANSFER FROM SHUTTLE TO ITOS ORBIT: A mechanism must be provided to transfer between the shuttle orbit and the ITOS orbit.
- D. LAUNCH COST IMPACT: Preliminary cost implications of using the shuttle transport method are examined.

### A. MISSION ANALYSIS

#### 1. General

The topics discussed as part of the mission analysis are:

- Viewing coverage
- Solar exposure
- Ground station contact
- Radiation environment
- Ground station operations

These areas have been examined for both the present ITOS and for advanced ITOS missions. For the present ITOS, the most cost-effective system was found to be a single satellite in a sun-synchronous orbit between 730 and 1000-n. miles altitude. For advanced ITOS, the cost-effective system is a single satellite in a sun-synchronous orbit between 600 and 1000-n. miles altitude.

#### 2. Viewing Coverage

An overriding requirement of the ITOS mission is that total global coverage will be provided every 12 hours or less. An additional requirement is that 10-percent data overlap be provided. Furthermore, the present ITOS sensors are not considered useful when the viewing zenith exceeds 65 degrees. Below an orbit altitude of 730 n. miles, these requirements cannot be met by a

single satellite. Figure A-1 shows the minimum single satellite altitude as a function of orbit inclination for a synoptic period of 12 hours, assuming a maximum allowed viewing zenith of 65 degrees. The two cases of "just contiguous coverage" and "10-percent overlap" are shown.

The addition of a second satellite, coplanar, and 180 degrees from the first, allows global coverage every 12 hours for orbit altitudes as low as 190 n. miles. However the expense of a second satellite can not be justified by any possible advantages of a lower orbit altitude.

For advanced ITOS missions, the maximum allowed viewing zenith will be 50 degrees, but 500-n. mile gaps will be allowed. Figure A-2 depicts the altitude-inclination envelope in which a 12-hour synoptic period can be achieved by a single satellite with the advanced ITOS constraints. For the advanced mission, the minimum single satellite orbit altitude is 500 n. miles, somewhat lower than for the present ITOS mission. The addition of a second satellite again lowers the orbit altitude necessary to achieve a 12-hour synoptic period. As stated previously, the addition of a second satellite is not cost-effective.

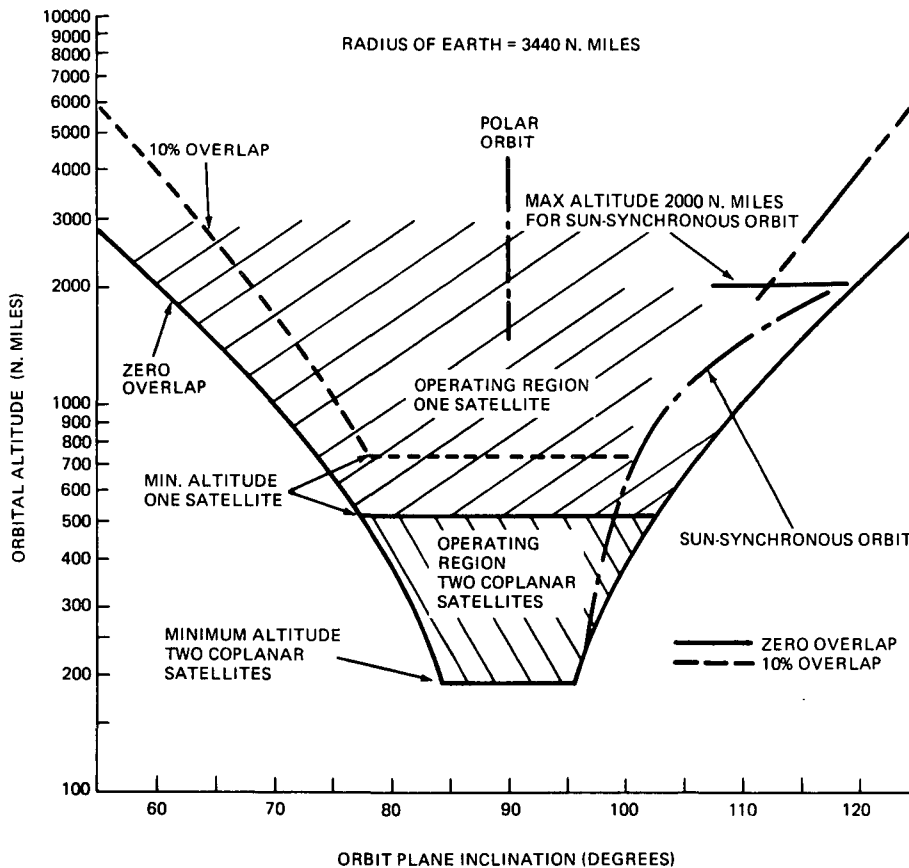


Figure A-1. Minimum Satellite Altitude for Global Coverage (Maximum Zenith Angle of 65 Degrees)

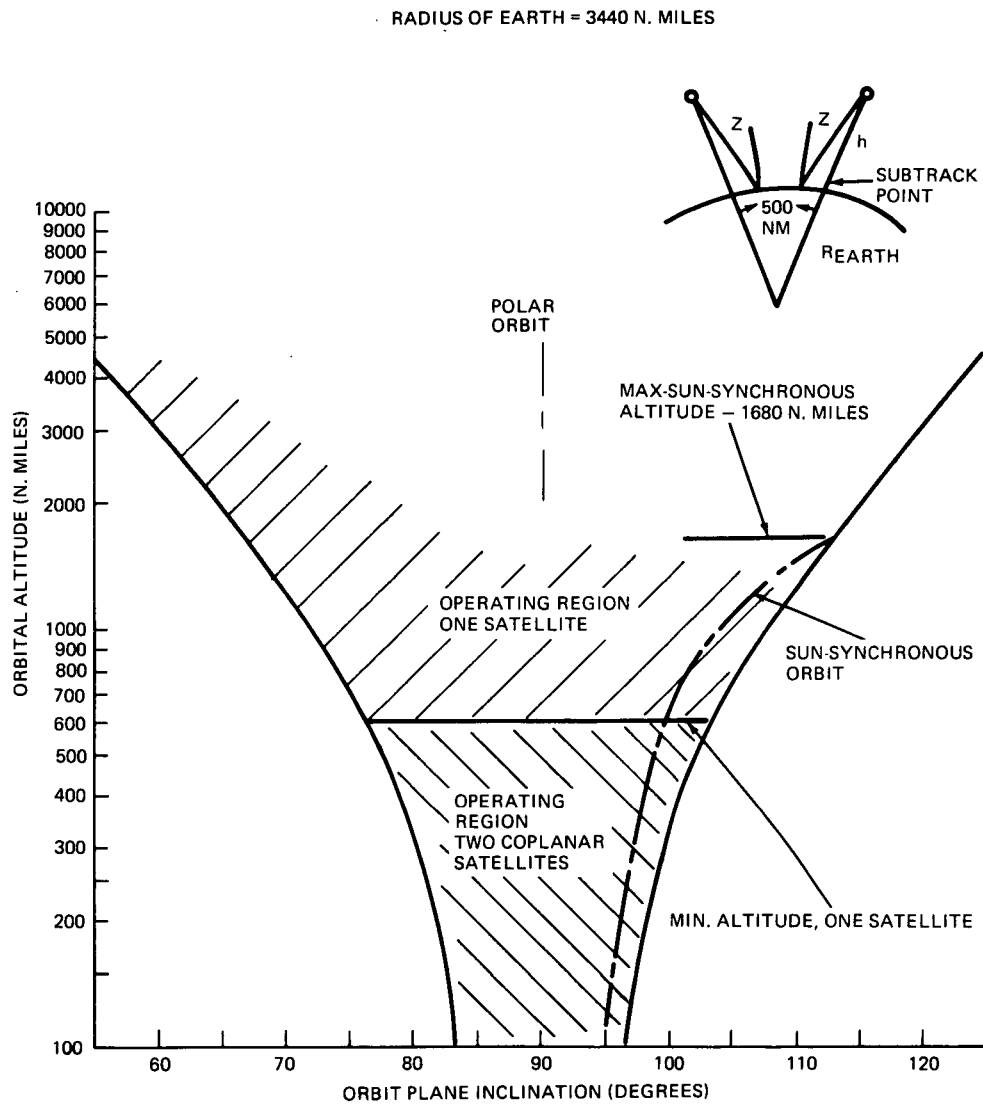


Figure A-2. Minimum Satellite Altitude for Global Coverage (Maximum Zenith Angle of 50 Degrees with 500-N. Mile Gap)

### 3. Solar Exposure

The current ITOS is launched into a sun-synchronous orbit. This orbit was selected to assure the least annual variation in spacecraft sun angle. As a result, annual fluctuations in solar energy conversion, duration of eclipse time, and spacecraft temperature are minimized. The impact of these minimized effects on spacecraft design leads to lower cost, weight, and complexity in that the ITOS spacecraft can be configured with a fixed, planar, solar array.

Other orbits such as posigrade and polar have been investigated for advanced ITOS and it can be clearly shown that either automatic sun tracking or body mounted solar arrays must be provided. In any event, satellite cost, complexity, and reliability are impacted. As a consequence, unless there are extensive advances in solar cell technology, the sun-synchronous mode will probably be continued for advanced ITOS.

Altitude and inclination combinations which produce sun-synchronous orbits are plotted on Figures A-1 and A-2. Possible ITOS orbits are those which lie on the sun-synchronous curve and within the 12-hour synoptic period envelope. Thus a lower and upper limit are established for the ITOS orbits. There is no cost advantage in going to higher than necessary orbits, and in practice orbits near the lower limit are chosen; 790 n. miles for the present ITOS and near 600 n. miles for the advanced ITOS. Once the final orbit altitude is chosen, the orbit inclination will be fixed as the inclination required to produce a sun-synchronous orbit at the specified altitude. The inclinations of interest fall between 100 and 109 degrees.

#### **4. Ground Station Contact Time**

Ground stations for ITOS are the Wallops Island and Alaska CDA\* stations. Contact is assumed to start and end 5 degrees above the horizon or terrain at either the Wallops or Alaska station.

For the minimum ground station contact time of 10 minutes (for adequate contact), the current ITOS in sun-synchronous orbit could have at most two successive orbits with marginal contact. Normally, a "missed orbit" condition will occur every 12 to 13 orbits so that adequate onboard storage for data playback during the following orbit is provided.

Ground station contact for polar orbits (above 800 n. miles) is 100 percent with the Alaska CDA station and approximately 60 percent of the time with Wallops. The number of missed orbits increases significantly with low inclination orbits. Consequently, the use of relay satellites would be necessary for certain posigrade orbits unless new ground stations are provided. This problem reduces as the orbit plane inclination approaches polar and as the altitude increases.

#### **5. Radiation Effects**

Degradation of the solar cells and the satellite sensors are functions of configuration, shielding, orbital characteristics and prevailing space radiation activity. The current ITOS with an orbit altitude of 790 n. miles sustains reasonably acceptable radiation effects on the order of 15 to 25-percent power degradation after one year.

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\*Command and Data Acquisition

By virtue of the increase in orbit altitude necessitated by viewing coverage considerations, satellites in posigrade orbits would be exposed to increased radiation (several orders of magnitude). This could cause significant increases in either the required power margin, shielding weight, or both, and, as a result, a reduction of cost effectiveness.

## **6. Ground Station Operations**

Although not a deciding factor, ground station operations and data handling are simplified and made less costly when the satellite ephemeris is time repeatable daily. This can be achieved by choosing the orbit period to result in an integral number of daily orbits as shown in the following tabulation:

Orbit altitude (n. miles)	475	675	915
Daily orbits	14	13	12

As a consequence, in order to meet viewing coverage constraints and yet simplify ground station operations, an ITOS orbit altitude of 915 n. miles should be selected. For the advanced ITOS profiling mission these same factors would lead to selecting an orbit altitude of 675 n. miles.

## **B. SHUTTLE PERFORMANCE**

### **1. General**

The discussion of shuttle performance includes a derivation of the shuttle payload-to-orbit capability from data in the "Space Shuttle Designer's Handbook."\* The handbook includes a curve of "Payload Vs. Orbital Inclination" (PVOI) for altitudes of 100 and 270 n. miles (see Figure A-3). This information has been utilized to generate shuttle payload-to-orbit capabilities for various altitudes and inclinations.

### **2. Payload-to-Orbit Derivations**

Characteristic velocities were calculated for combinations of altitude and inclination on the PVOI curve of the Designer's Handbook. The calculated characteristic velocities corresponding to each of these combinations of altitude and inclination were paired with the appropriate payloads on the PVOI curve to produce the graph shown in Figure A-4. This plot of characteristic velocity versus payload was then used to generate the payload-to-orbit curves of Figure A-5. The following paragraphs describe the characteristic velocity calculation which was modified from the standard approach to reflect the requirement of returning the shuttle to earth.

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\*Space Shuttle Designer's Handbook (Draft), provided by A. Kampinsky, Advanced Plans Staff, NASA.

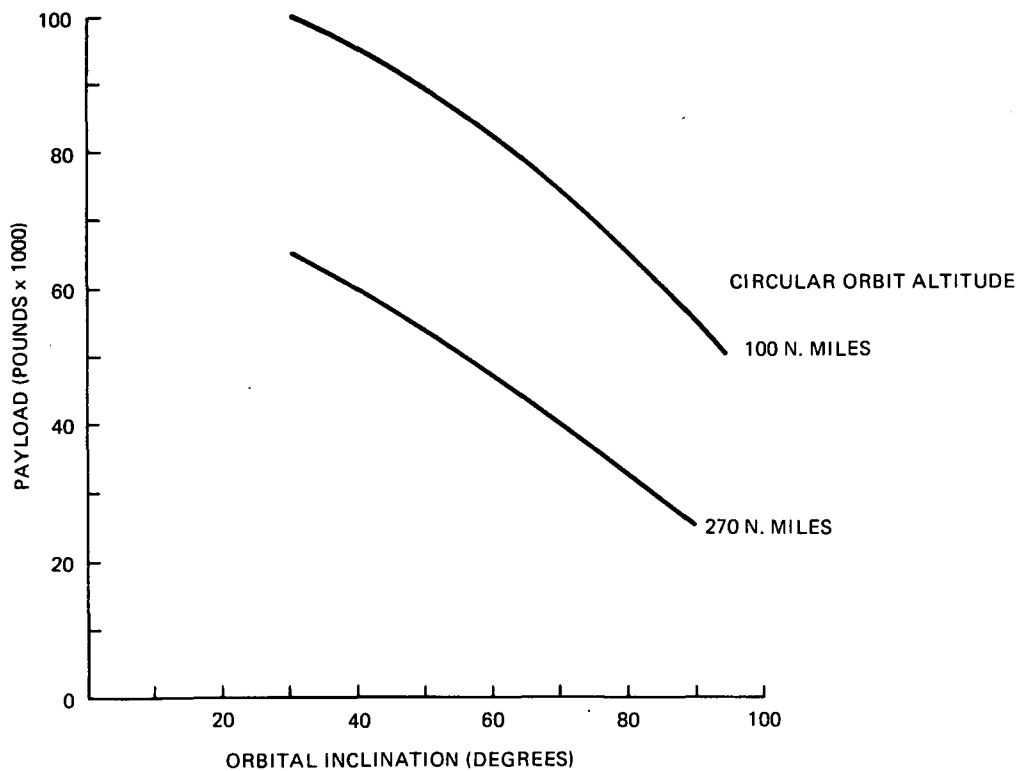


Figure A-3. Payload vs. Orbital Inclination

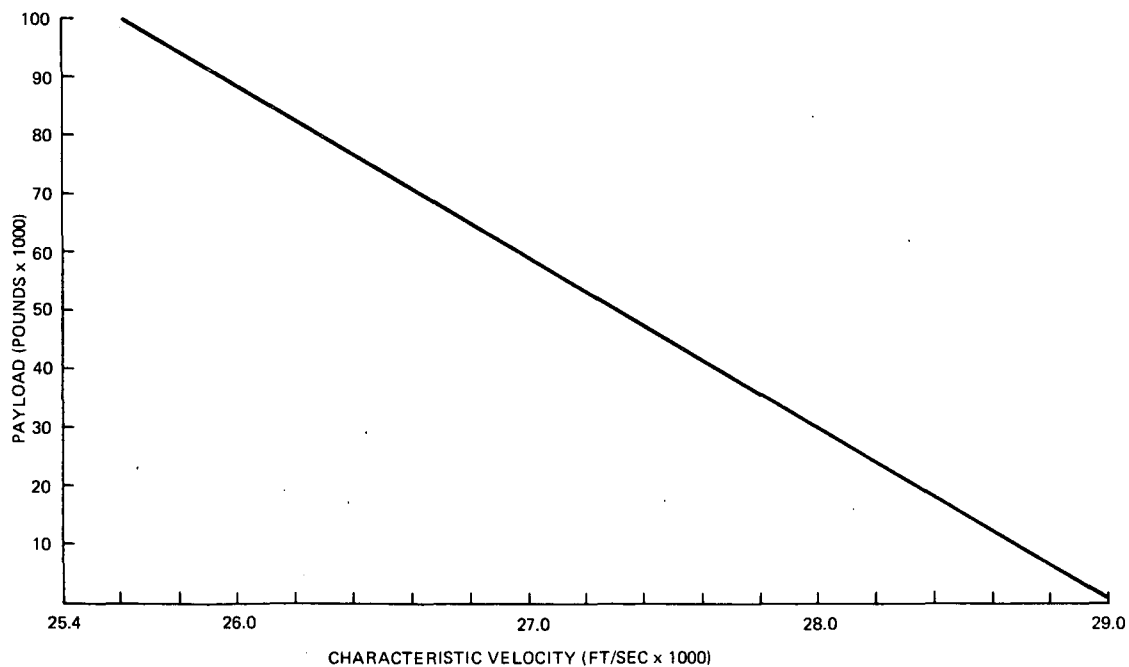


Figure A-4. Characteristic Velocity vs. Payload

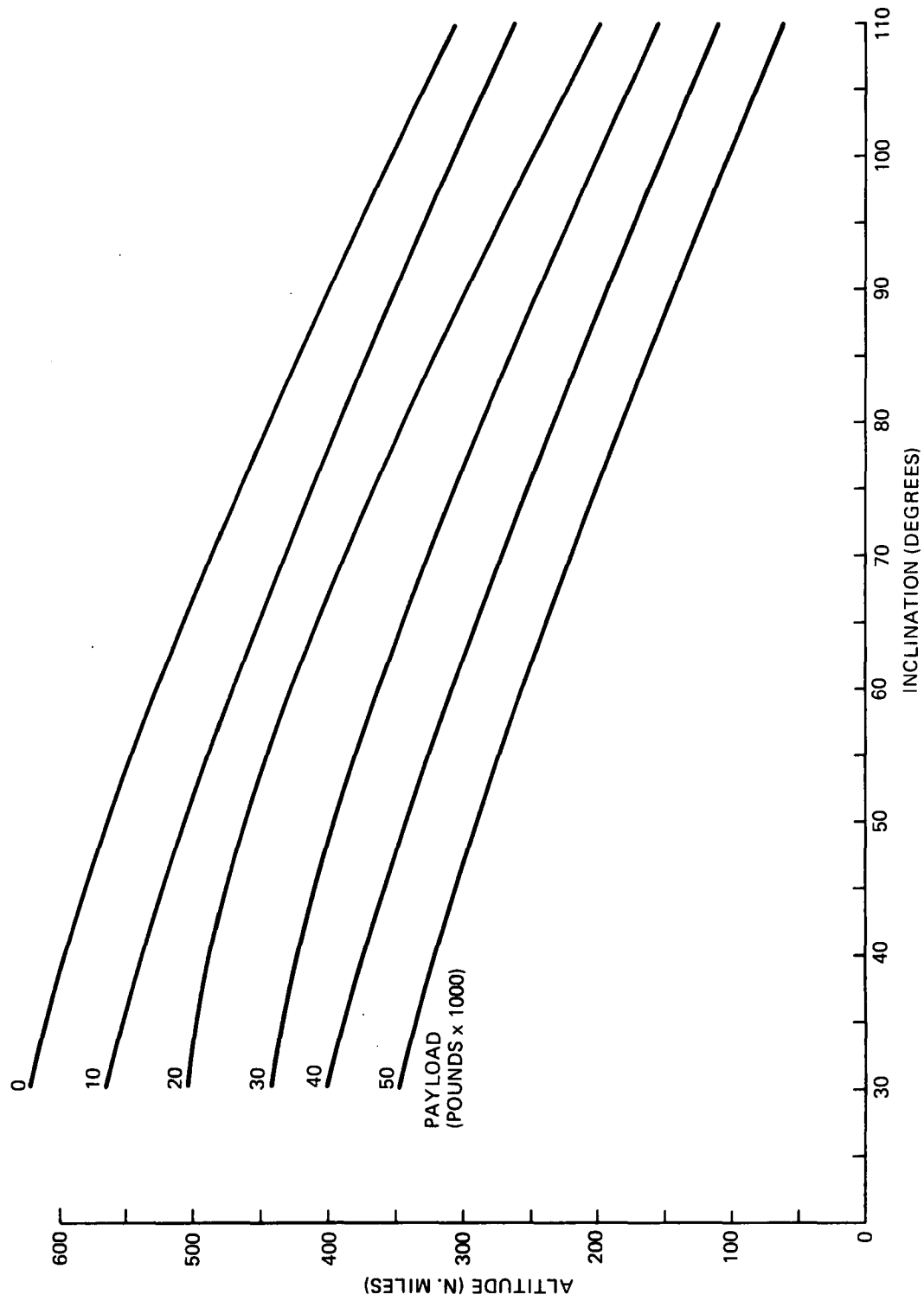


Figure A-5. Payload-to-Orbit Capability

The characteristic velocities calculated have been those required to bring the shuttle to a particular circular orbit altitude and inclination and then return it to a 100 n. mile circular orbit. This characteristic velocity, which will be called  $V_{alt \& ret}$ , is taken to be:

$$V_{alt \& ret} = V_{ch \ alt} + V_{lp} + (V_{ch \ alt} - V_{ch \ 100}) \quad (A-1)$$

where

$V_{ch \ alt}$  is the characteristic velocity required to reach a orbit at a particular altitude, with no return.

$V_{lp}$  is the launch penalty required to reach the inclination desired, assuming an Eastern Test Range launch.

It should be noted that several assumptions are implicit in equation A-1. The first is that the shuttle need not be returned to any specific inclination in order to deorbit. This assumption is based on the 1500-n. mile cross-range capability of the shuttle, which will make an acceptable landing field available from any inclination. However, the velocity requirement to fly to a specific altitude (above 100 n. miles) is taken to include the  $\Delta v$  requirement to return the vehicle to a 100 n. mile altitude. As the vehicle goes higher, the  $\Delta v$  to deorbit becomes greater and the additional  $\Delta v$  to deorbit contributes to reducing the payload.

The second assumption is that the PVOI curves of the Designer's Handbook are based on the vehicle deorbiting from 100 n. miles; that is, the payload penalties for return from this altitude are already included. This assumption was checked by the consistency of the characteristic velocity vs. payload curve obtained from the PVOI curves for the two different altitudes.

It has also been assumed that the shuttle actually has the restart capability necessary to perform the burn profile required by the transfers implied in Figure A-6. All the altitude transfers are assumed to be Hohmann transfers.

### 3. Impact on ITOS

From Figure A-5 it can be seen that the shuttle cannot directly inject a payload into the ITOS orbit range of 600 to 1000-n. mile altitude with inclinations between 100 and 105 degrees. Paragraph C of this Appendix discusses the requirements for going from a possible shuttle orbit to possible ITOS orbits.

## C. TRANSFER FROM SHUTTLE TO ITOS ORBIT

### 1. General

From the preceding two sections it is clear that the shuttle cannot deliver payloads directly to the desired ITOS orbits. This section will discuss



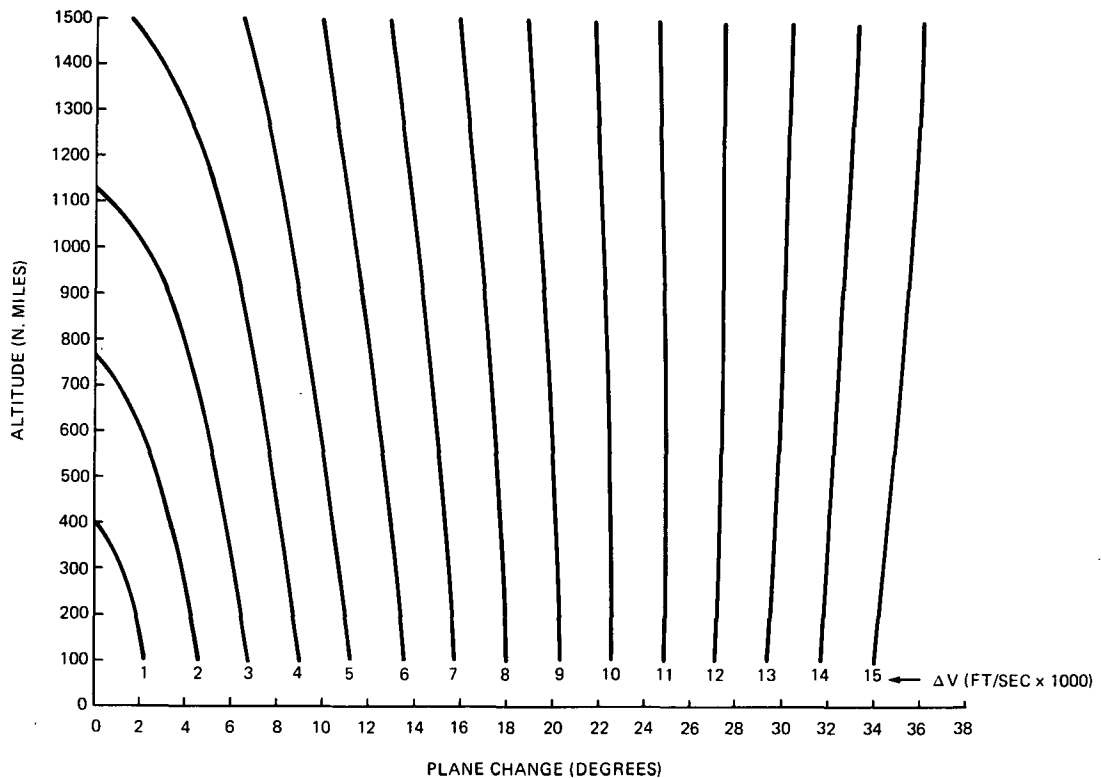


Figure A-6. Plane Change and Altitude Combinations vs.  $\Delta v$  (Initial Altitude of 100-N. Miles)

the velocity change requirements for transferring between the shuttle and ITOS orbits; propulsion systems to perform the orbit transfers will also be discussed.

## 2. Velocity Change Requirements

The velocity changes required for two cases of orbit transfer are treated parametrically. For Case I, a satellite is assumed to be in a circular orbit of specified altitude and inclination; the velocity change required to move the satellite to a circular orbit at a new altitude and inclination is calculated. The calculation is then particularized by choosing orbits with sun-synchronous inclination. For Case II a satellite is assumed to be in a circular orbit at a specified altitude, inclination and right ascension of the ascending node; the velocity change required to move the right ascension of the ascending node while keeping the altitude and inclination constant is calculated.

### a. CASE I: TRANSFER FROM A CIRCULAR ORBIT TO A HIGHER CIRCULAR ORBIT HAVING SUN-SYNCHRONOUS INCLINATION

The desired transfer can be accomplished with two impulsive maneuvers. The first impulse is applied to put the vehicle in a Hohmann transfer ellipse

with the perigee at the initial altitude and the apogee at the final altitude. The velocity change required for the initial impulse is:

$$\begin{aligned}\Delta v_1 &= v_p - v_{cl} \\ &= \sqrt{\frac{2 GMR_a}{(R_a + R_p) R_p}} - \sqrt{\frac{GM}{R_p}}\end{aligned}\tag{A-2}$$

where

$v_p$  is the velocity at perigee of the transfer ellipse

$v_{cl}$  is the circular velocity at initial altitude

$G$  is the universal gravitational constant

$M$  is the mass of the earth

$R_p$  is the distance from center of the earth to perigee of the transfer ellipse

$R_a$  is the distance from center of the earth to apogee of the transfer ellipse

The second impulse is applied at the apogee of the transfer ellipse with a magnitude and direction which will simultaneously circularize at the final altitude and produce the desired inclination. The magnitude of the second velocity change is:

$$\begin{aligned}\Delta v_2 &= \sqrt{v_a^2 + v_{c2}^2 - 2v_a v_{c2} \cos \theta} \\ &= \sqrt{\frac{2GMR_p}{(R_a + R_p) R_a} + \frac{GM}{R_a} - 2\sqrt{\frac{2GMR_a}{(R_a + R_p) R_a}} \sqrt{\frac{GM}{R_a}} \cos \theta} \\ &= \sqrt{\frac{GM}{R_a}} \sqrt{1 + \frac{2R_p}{R_a + R_p} - 2 \cos \theta} \sqrt{\frac{2R_p}{R_a + R_p}}\end{aligned}\tag{A-3}$$

where

$v_a$  is the velocity at apogee of the transfer ellipse

$v_{c2}$  is the circular velocity at the final altitude

$\theta$  is the difference between the final and initial inclination

The total velocity change required for the Case I transfer is the sum of  $\Delta v_1$  and  $\Delta v_2$ .

$$\Delta v_T = \Delta v_1 + \Delta v_2 \quad (A-4)$$

Equation A-4 was solved for  $\theta$  as a function of  $\Delta v_T$ ,  $R_a$ , and  $R_p$ . A computer program was generated which calculated  $\theta$  for input values of  $\Delta v_T$ ,  $R_a$ , and  $R_p$ . The results are plotted in Figure A-6 for an initial altitude of 100 n. miles and in Figure A-7 for an initial altitude of 270 n. miles.

Equation A-4 was then evaluated for  $\theta$  equal to the difference between sun-synchronous inclination at the final altitude and the initial inclination.

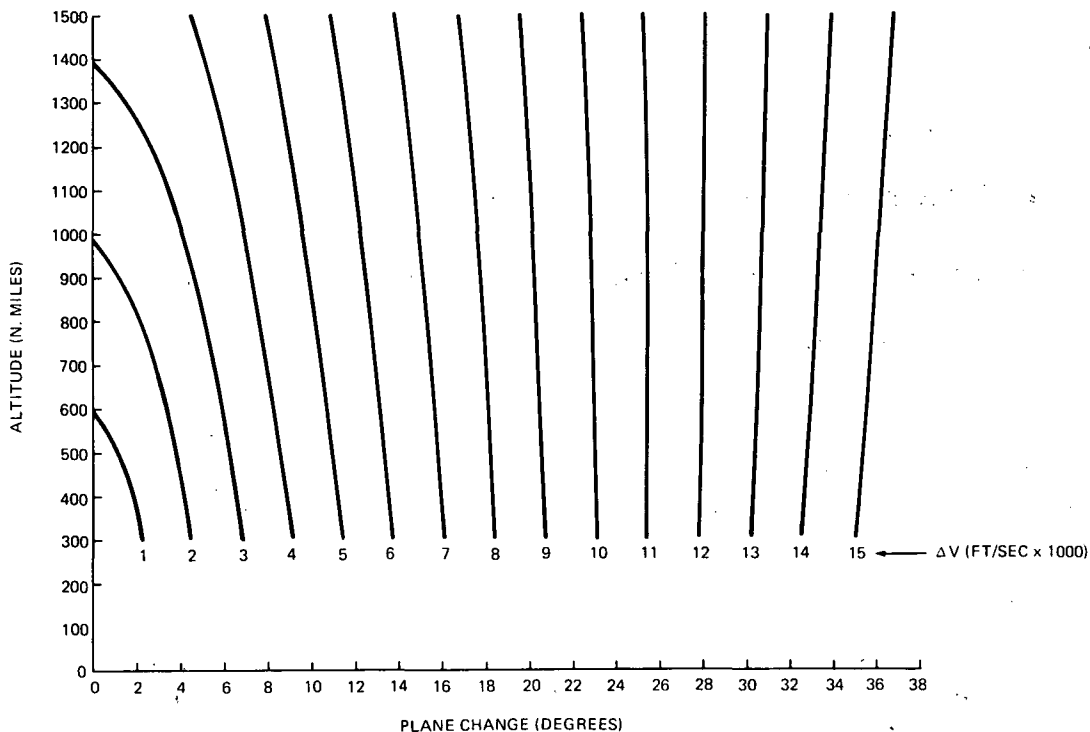


Figure A-7. Plane Change and Altitude Combinations vs.  $\Delta v$   
(Initial Altitude of 270 N. Miles)

The total velocity change was evaluated for transfers between initial altitudes of 100, 270, 500 and 800 n. miles at inclinations of 28.5, 55, 70, and 90 degrees and final altitudes of 800, 1200 and 1500 n. miles with sun-synchronous inclination. The results are plotted parametrically in Figure A-8.

b. CASE II: CHANGING THE RIGHT ASCENSION OF THE ASCENDING NODE OF A CIRCULAR ORBIT WHILE KEEPING THE INCLINATION FIXED

The Case II transfer can be accomplished with a single impulse. Figure A-9 shows the relative geometry between the initial and final orbits. The impulse is performed at  $\theta$  to produce the desired transfer. The velocity change required is:

$$\Delta v = \sqrt{v_1^2 + v_2^2 - 2v_1v_2 \cos \theta} \quad (A-5)$$

where

$v_1 = v_2 = v_c$ , the circular velocity at the given altitude

$\theta$  is the angle between the initial and final orbit.

Referring to Figure A-9 and using spherical trigonometry,  $\cos \theta$  is found to be:

$$\cos \theta = -\cos i \cos (\pi - i) + \sin i \sin (\pi - i) \cos (\Delta \Omega)$$

where

$i$  is the specified inclination

$\Omega$  is the right ascension of the ascending node.

Substituting the relation for  $\cos \theta$  in equation A-5 and simplifying leads to:

$$\Delta v = 2v_c \sin i \sin \frac{\Delta \Omega}{2}$$

The parametric plots of Figure A-10 show the velocity change required at altitudes of 270, 800, 1200 and 1500 n. miles with inclinations of 28.5, 55, 70 and 90 degrees to produce node rotations of 10, 20, 30, 40, 50, 60, 70, 80 and 90 degrees.

### 3. Payload/Orbital Transfer/ $\Delta v$

From Figure A-8 we see that unless the shuttle is launched to a near-polar orbit, the velocity change requirement to transfer the ITOS vehicle into its final sun-synchronous orbit becomes prohibitive. From a polar orbit with 270-n. mile altitude, the  $\Delta v$  requirement to reach a typical ITOS orbit is approximately 6000 ft/sec.

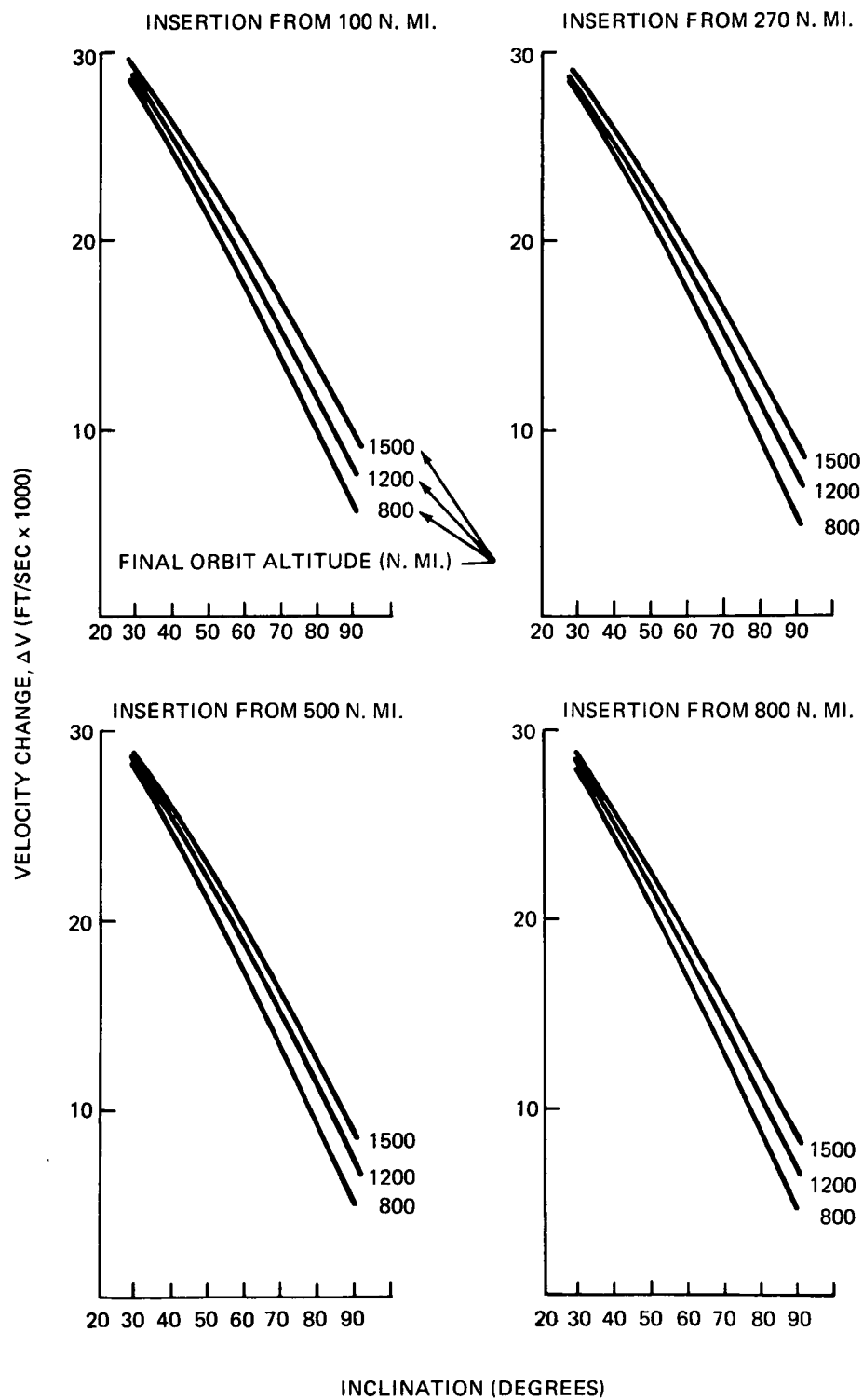


Figure A-8. Velocity Change Requirement for Transfer from an Initial Altitude and Inclination to a Final Altitude Having Sun-Synchronous Inclination

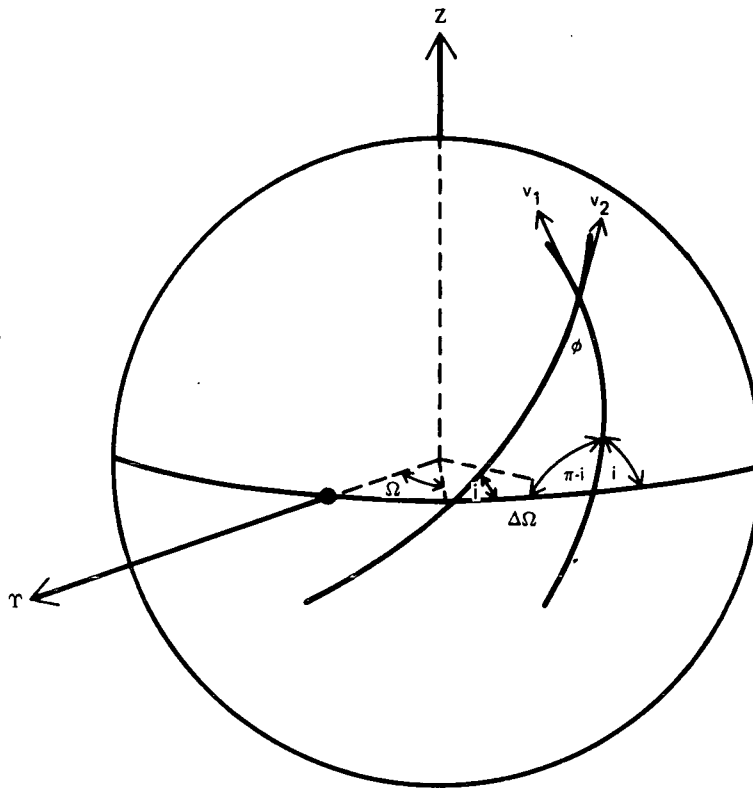


Figure A-9. Relationship Between Initial and Final Case II Orbits

From the same initial altitude, but with a 55-degree inclination, the  $\Delta v$  requirement to reach a typical ITOS orbit is between 19,000 and 20,000 ft/sec. A  $\Delta v$  of 6,000 ft/sec is manageable, but 20,000 ft/sec is unreasonable. Unfortunately, as orbit inclination increases, the payload capability of the shuttle rapidly decreases (see Figure A-5). For example, the shuttle can inject the maximum 50,000 pounds of payload into a 270-n. mile, 55-degree orbit, but only 25,000 pounds into a 270-n. mile, 90-degree orbit. Launching the shuttle to near-polar orbits decreases both the shuttle capacity and the  $\Delta v$  required for transfer to the final ITOS mission orbit. This decrease in shuttle capacity is acceptable, and indeed preferable to the high  $\Delta v$  requirement associated with ITOS orbit transfers from lower inclination shuttle orbits. Further examination of the shuttle payload-to-orbit curves reveals that if the shuttle is launched to an orbit altitude of 100 n. miles, 50,000 pounds of payload can be delivered to near-polar orbits. Figure A-8 shows that there is very little difference between the  $\Delta v$  requirements to transfer from a 270-n. mile, 90-degree or 100-n. mile, 90-degree orbit to an 800-n. mile sun-synchronous orbit. If the shuttle operates at a lower initial orbit altitude, its payload capacity can be greatly increased with only a small change in the  $\Delta v$  required to bring the payload from the shuttle orbit to the final ITOS orbit.

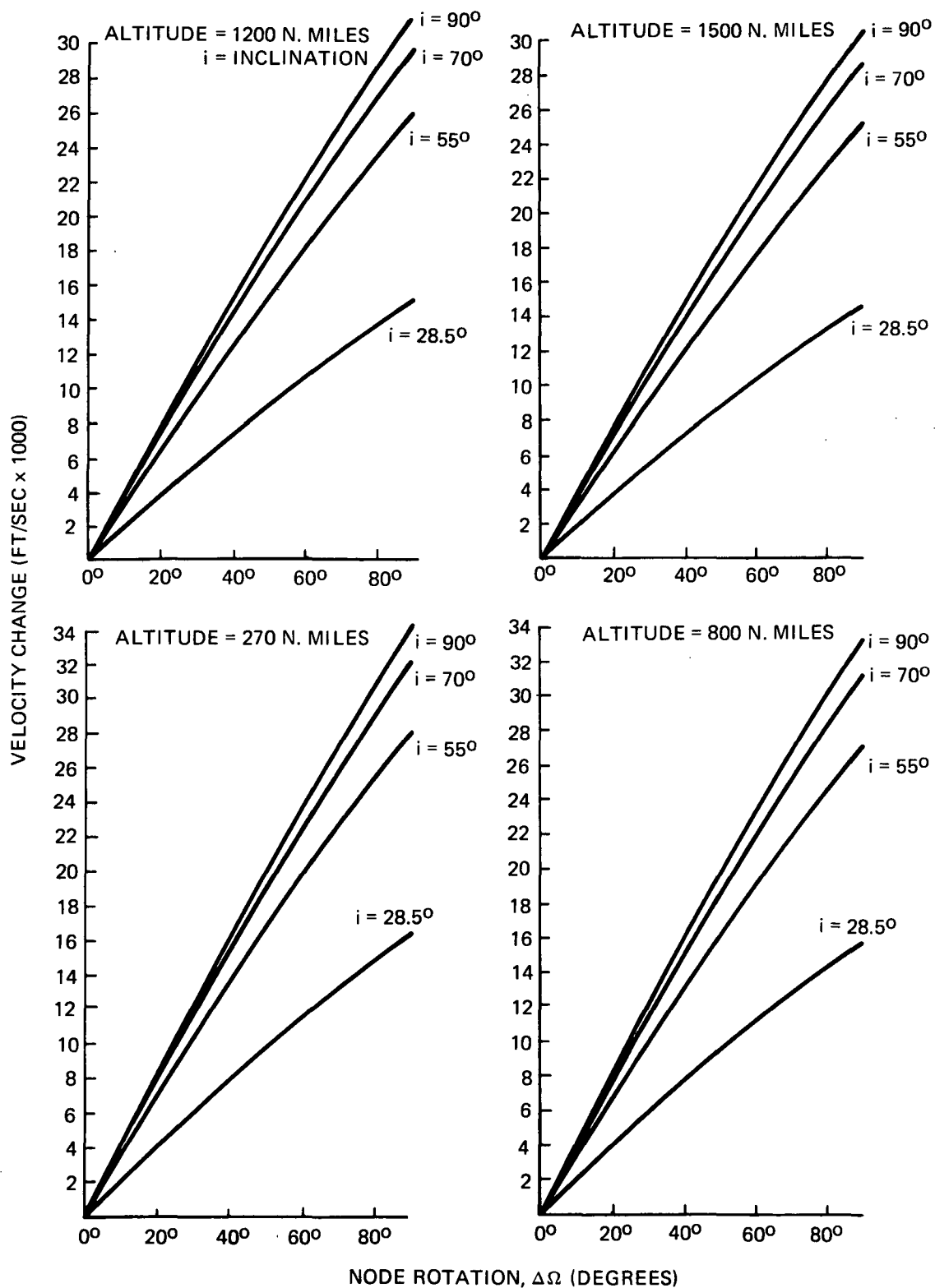


Figure A-10. Velocity Change Requirements, Case II Transfers

#### 4. Propulsion Systems

An auxiliary propulsion system will be necessary to transfer the ITOS vehicle between the shuttle orbit and the ITOS mission orbit. Two types of auxiliary propulsion systems were examined, an unmanned expendable booster and a manned reusable tug.

##### a. EXPENDABLE BOOSTER

The expendable booster is conceived of as a propulsion package to be brought up in the shuttle with the ITOS vehicle; it will be integrated with the ITOS vehicle and perform a perigee burn and an apogee burn to place ITOS in its final orbit. A number of two-stage booster combinations were considered, each designed to satisfy a specific range of velocity change requirements. Figure A-11 depicts the ITOS spacecraft with expendable booster package in the shuttle cargo bay.

Table A-1 is a summary of the expendable booster packages considered. The "RCA Kick I" package, a two-stage vehicle, is configured to satisfy small  $\Delta v$  requirements (less than 2500 ft/sec); it contains the required guidance and control. A similar "RCA Kick II", will meet intermediate  $\Delta v$  requirements (2500 to 7500 ft/sec), and an "RCA Kick III", will provide still larger  $\Delta v$ 's (7500 to 10,000 ft/sec). Finally, a Delta/TE-364 two-stage booster will satisfy very large  $\Delta v$  requirements (10,000 to 25,000 ft/sec). The Delta/TE-364 combination is thought to be at the limit of cost effectiveness.

TABLE A-1. SUMMARY OF EXPENDABLE BOOSTER PACKAGES

Expendable Booster	Booster Weight (pounds)	Booster $\Delta v$ Capability (ft/sec)
RCA Kick I	685	2500
RCA Kick II	2090	2500 to 7500
RCA Kick III	3500	7500 to 10,000
Delta TE-364	13,961	10,000 to 25,000



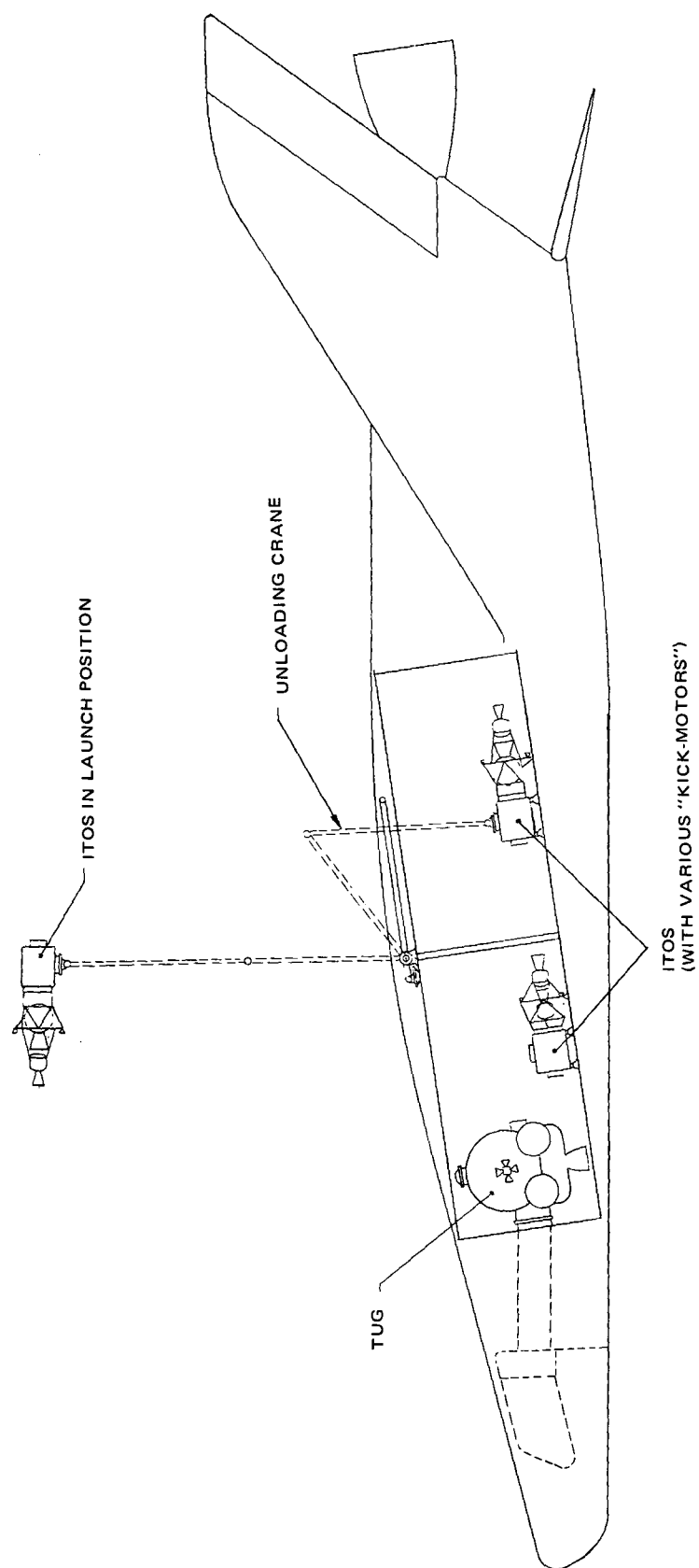


Figure A-11. Spacecraft/Kick-Motor Sequence

b. REUSABLE TUG

An alternative to the expendable booster is a reusable tug. The tug, as defined in Section II, paragraph A6, is designed specifically for transferring small satellites from one orbit to another. The assumptions concerning the tug are:

- (1) The tug will be built and operable when the shuttle is available as a transport.
- (2) The tug will weigh approximately 3,000 pounds.
- (3) The tug maximum fuel capacity will be 7,500 pounds.
- (4) The tug will use a fuel with a specific impulse of 300 sec.

When the tug is considered as an orbit transfer vehicle two possible situations arise:

- (1) The tug, its fuel, and the ITOS spacecraft are all carried into orbit as part of a shuttle payload.
- (2) The tug is already in the shuttle orbit; the shuttle carries the ITOS spacecraft and fuel for the tug.

Both of these possibilities are considered in the section on launch costs, paragraph D of this Appendix.

D. LAUNCH COST IMPACT

1. General

Two methods for using the shuttle to transport the ITOS to its mission orbit were described in the previous section. The cost of these transport modes are now compared with the present Delta Thor launch.

Launching ITOS on a Delta Thor will cost \$4 million. The anticipated cost of launching the space shuttle is \$5 million. It is assumed that the shuttle will carry several payloads, and that the launch cost of \$5 million will be pro-rated over these payloads. If the cost of the auxiliary propulsion system required to transfer the ITOS spacecraft to its final orbit, summed with the pro-rated cost of the shuttle weight and volume capacity required by the ITOS mission, is below the Delta Thor launch cost of \$4 million, then using the shuttle transport mode will be considered cost effective. The cost estimates made in this study are based on the assumption that the shuttle is launched at or near full capacity. In addition, it is assumed that no change in right ascension is required to reach the ITOS orbit. The cost estimates calculated for the two types of auxiliary propulsion systems are presented in paragraphs D2 and D3, which follow.

## 2. Expendable Booster

For launches which use one of the expendable booster systems to inject ITOS into its final orbit, the launch cost will be the price of the expendable booster plus the cost of lifting the ITOS spacecraft and the expendable booster into the shuttle orbit. Estimates for this type of launch are summarized in Table A-2. This table is applicable to shuttle launches into 270-n. mile orbits and transfers to 790-n. mile, 102-degree sun-synchronous orbits from the inclinations indicated. In all cases the weight of the ITOS spacecraft was assumed to be 1000 pounds.

TABLE A-2. COST ESTIMATES FOR LAUNCHES WITH EXPENDABLE BOOSTERS

Shuttle Orbit Inclination	102°	90°	80°	56°
Shuttle Altitude (n.mile)	270	270	270	270
Plane Change to Sun-Synchronous Orbit	0°	12°	22°	46°
Available Shuttle Payload Capacity (lb.)	18,000	26,000	33,000	50,000
Payload Delivery Cost (dollars/lb)	278	192	151	100
Perigee $\Delta v$ (ft/sec)	800	800	800	800
Apogee $\Delta v$ (ft/sec)	900	5,600	8,700	18,700
Applicable Booster	RCA Kick I	RCA Kick II	RCA Kick III	Delta TE-364
Weight of Booster (lb)	685	2090	3500	13,961
Booster Cost (\$ million)	0.60	0.67	0.75	2.5
Total Payload Weight (with 1000-lb ITOS)	1685	3090	4500	14,961
Payload Delivery Cost (\$ million)	0.48	0.59	0.68	1.5
Total Launch Cost (\$ million)	1.08	1.26	1.43	4.0

### 3. Reusable Tug

When the reusable tug performs the final ITOS injection, two possibilities occur. The tug may already be in the shuttle orbit; in this case only propellants for the tug need be lifted into orbit with the ITOS spacecraft. If there is no tug waiting in the shuttle orbit, then the tug must be launched along with its propellants and the ITOS spacecraft. In both cases the amount of propellant required will be enough to bring the tug/ITOS combination to the ITOS mission orbit and then return the tug to the shuttle orbit.

Propellant requirements for the tug were calculated from the equation

$$W_{\text{prop}} = W_d \left[ \exp (\Delta v / I_{\text{sp}} g_c) - 1 \right] \quad (\text{A-6})$$

where

$W_d$  is the weight of the propulsion system plus ITOS, less weight of the propellants to be consumed

$\Delta v$  is the velocity change to be performed

$I_{\text{sp}}$  is the specific impulse of the propellant

$g_c$  is the acceleration due to gravity at the surface of the earth

Propellant costs were considered negligible compared to the cost of lifting the propellant into orbit.

The cost estimates for using the tug in either of its two possible modes are presented in Figure A-12. The cost estimates were made for the case of no plane change and for a 12-degree plane change. The Delta Thor cost is also listed for comparison.

### 4. Conclusions

The calculations made here are of a preliminary and approximate nature; however, the results indicate a potential launch cost saving on the order of 50 percent if the shuttle transport mode were made available for ITOS launches.

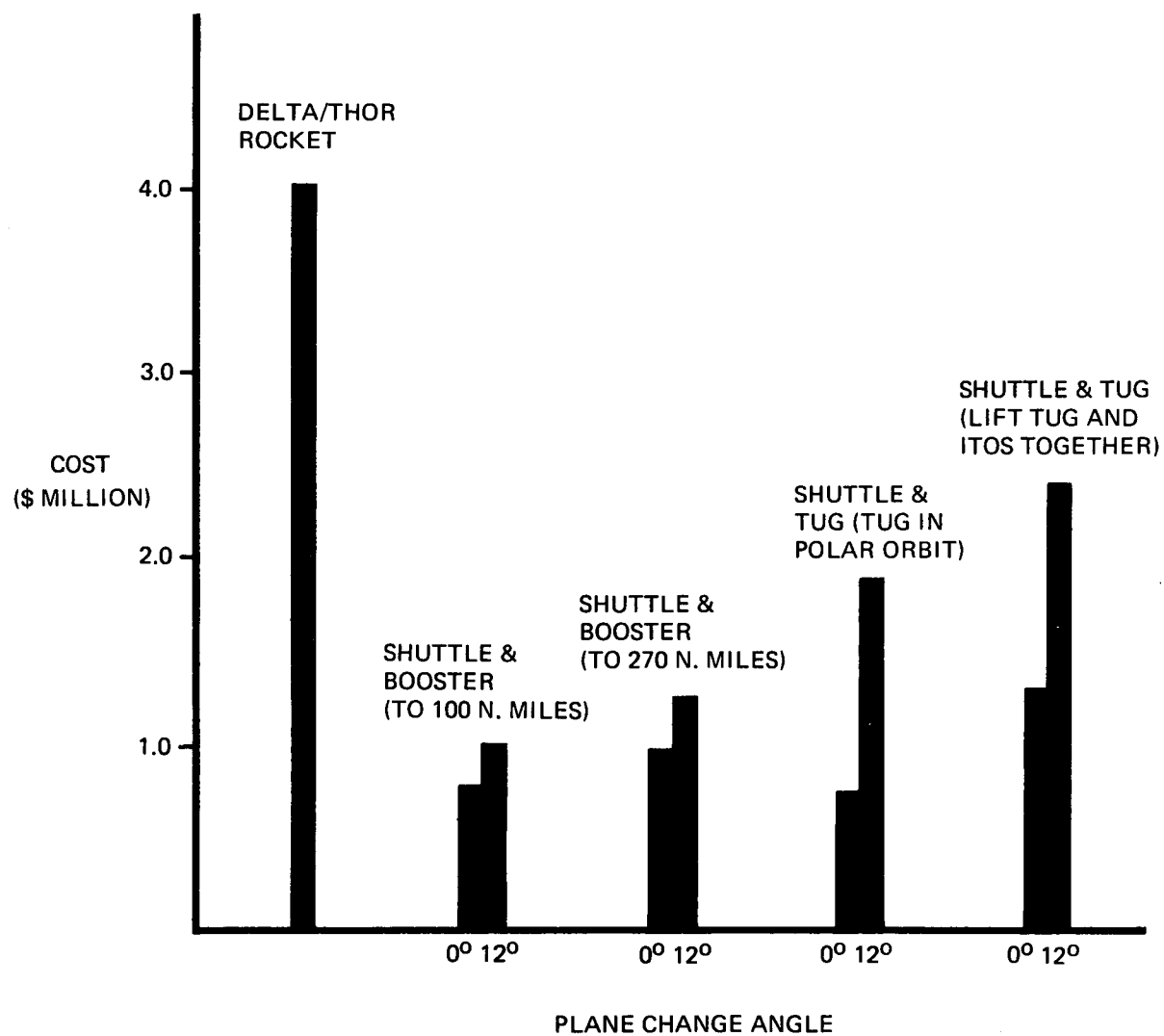


Figure A-12. Injection Costs to Attain 102-Degree, 790-N. Mile Orbit